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# RESEARCH MEMORANDUM

ALTITUDE EVALUATION OF SEVERAL AFTERBURNER DESIGN  
VARIABLES ON A J47-GE-17 TURBOJET ENGINE

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RESEARCH MEMORANDUM

## ALTITUDE EVALUATION OF SEVERAL AFTERBURNER DESIGN VARIABLES

## ON A J47-GE-17 TURBOJET ENGINE

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## SUMMARY

An investigation was conducted in an NACA altitude chamber to evaluate the effectiveness of turbine-outlet gas-straightening vanes and vortex generators, fuel distribution modifications, and afterburner shell cooling as means of improving afterburner performance. Installation of the turbine-outlet gas-straightening vanes and vortex generators resulted in a lower total-pressure loss through the afterburner and an altered air-flow profile at the afterburner inlet. Therefore, it was necessary to modify the fuel distribution to provide a good fuel-air environment at the flame holder. Another result of the installation of the turbine-outlet gas-straightening vanes and vortex generators was the reduction of the afterburner shell temperature by approximately 100° R. An additional 100° R reduction in the afterburner shell temperature was obtained with a ceramic-coated corrugated liner.

The best afterburner configuration of this investigation incorporated the turbine-outlet gas-straightening vanes and vortex generators, the ceramic-coated corrugated liner, and a fuel distribution that provided a good fuel-air environment at the flame holder. This configuration had higher afterburner combustion efficiencies than the original configuration, and the altitude limit was in excess of 54,000 feet. At low altitudes, the operation of this configuration was not limited by afterburner shell temperatures.

## INTRODUCTION

Current and proposed military aircraft require thrust in addition to normal engine thrust for take-off, climb, and high speed at high altitudes. Using an afterburner is one method of meeting these require-

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ments for additional thrust. Accordingly, the NACA is actively conducting several related afterburner programs.

The investigation reported herein presents information on design factors and modifications of the production afterburner for the J47-GE-17 turbojet engine designed for medium temperature operation. The present report is concerned only with the afterburner performance and operating characteristics. Altitude-starting characteristics of two of the configurations in this report are discussed in reference 1. This investigation was conducted with an engine equipped with a variable-area exhaust nozzle and operated over a range of simulated flight conditions in a 10-foot-diameter altitude test chamber at the NACA Lewis laboratory.

The initial configuration was developed by the manufacturer from a previous production afterburner by incorporating design modifications indicated by investigations reported in references 2 to 5. In the present investigation, attention was focused on three primary factors in order to improve the performance and operating limits: (1) turbine-outlet gas whirl, (2) matching of the fuel distribution with the mass-flow distribution of the turbine gases, and (3) afterburner shell cooling. Previous investigations (ref. 6) indicate that turbine-outlet gas whirl can have a detrimental effect on the performance characteristics of the afterburner. Accordingly, the effect of turbine-outlet gas-straightening vanes and vortex generators on performance was evaluated in this investigation. Several fuel-spray-bar configurations were also evaluated in order to provide satisfactory matching of the fuel distribution with the mass-flow distribution. It was desirable to evaluate methods of afterburner shell cooling, since maximum thrust can be limited by afterburner shell temperatures. Two methods of cooling the afterburner shell were investigated: (1) the incorporation of a method of fuel distribution that provided a lean fuel-air mixture near the outer shell and (2) installation of a ceramic-coated corrugated liner in the burner section.

The performance with these modifications incorporated is presented over a range of altitudes up to 50,000 feet. Data are presented in tabular and graphical form to show the effects on afterburner performance of turbine-outlet gas-straightening vanes and vortex generators, fuel distribution, and a ceramic-coated corrugated liner and to illustrate the effect of variations in flight conditions on one of the best afterburner configurations.

## APPARATUS AND INSTRUMENTATION

## Installation

The engine was installed in an altitude chamber that is 10 feet in diameter and 60 feet long (fig. 1). A honeycomb installed in the chamber upstream of the test section straightened and smoothed the flow of the inlet air. A forward bulkhead, which incorporated a labyrinth seal around the forward end of the engine, separated the engine-inlet air from the exhaust and provided a means of maintaining a pressure difference across the engine. A 14-inch butterfly valve in the forward bulkhead provided cooling air for the engine compartment, and a rear bulkhead prevented recirculation of exhaust gases about the engine. The exhaust gas from the jet nozzle was discharged into an exhaust diffuser to recover some of the kinetic energy of the jet and thus to extend the capacity of the exhaust system. The combustion in the afterburner was observed through a periscope located directly behind the engine.

## Engine

A J47-GE-17 afterburning turbojet engine was used in this investigation. The engine has a static sea-level thrust rating of 5420 pounds without afterburning at the rated engine speed of 7950 rpm and a turbine-discharge temperature of 1760° R for an inlet air temperature of 519° R. At this operating condition, the air flow is 104 pounds per second. The over-all length of the engine and afterburner is approximately 228 inches, and the maximum diameter is 41 inches. The dry weight of the engine and afterburner, including the electronic control and airframe mounted components, is 3553 pounds. The electronic control (described in ref. 1) controls the engine speed by regulating engine fuel flow and controls the turbine-outlet temperature by regulating the exhaust-nozzle area.

## Afterburner Assembly

A diagram of the afterburner assembly is shown in figure 2. The following were common to all configurations: conical diffuser, two-ring V flame holder mounted by struts from the inner body, converging conical burning section, and variable-area clamshell exhaust nozzle. The conical inner body, mounted from the outer shell by four tubular rods, contained the fuel manifolds and a depressed flame seat in the downstream end. Fuel was supplied to the afterburner by an air-turbine fuel pump driven by compressor bleed air.

Modifications to the afterburner, which were incorporated in six configurations, are listed in table I. The fuel-distribution patterns and spray-bar designs are presented in figure 3. A photograph of the turbine-outlet gas-straightening vanes and vortex generators is shown in figure 4. The ceramic-coated corrugated liner is shown installed in the afterburner section in figure 5. The ceramic used was National Bureau of Standards number A418, primarily composed of chromium oxide.

The six configurations are grouped as to purposes of the modifications. The first group of modifications was selected to show the effect of straightening vanes on engine and afterburner performance. This group included the manufacturer's original configuration A and configuration B, which was made by incorporating in configuration A straightening vanes and vortex generators at the turbine outlet and a 4-inch shorter fuel mixing length (the radial fuel distribution was equivalent to that of configuration A). The shorter fuel mixing length was a result of moving the spray bars downstream of the inner-cone supporting struts to allow equidistant circumferential spacing of the bars.

The second group of configurations was selected to illustrate the effect of modified fuel distribution. These configurations were geometrically similar, all having straightening vanes, vortex generators, and corrugated liners. Configuration C had a radial fuel distribution equivalent to the original configuration A. In configurations D and E, the fuel distribution was modified to compensate for the shift in mass-flow profile caused by the installation of the straightening vanes and vortex generators. This modification was based on the total-pressure profile at the diffuser outlet. The spray bar for configuration D had equally spaced holes of varied diameters, while the bar for configuration E had varied spacing of equally sized holes. The bar for configuration E, designed by the manufacturer, was a uniorifice bar with a metering orifice at the inlet to the bar that permitted the use of larger spray holes to prevent plugging by foreign material in the fuel.

The third group of modifications was selected to evaluate several methods of cooling the outer shell. Configurations A, B, C, and F are compared. Configuration A was the original configuration already described. Configuration F had a fuel distribution that was lean near the outer shell but was otherwise the same as configuration A. Configuration B differed from configuration A in that turbine-outlet gas-straightening vanes and vortex generators were added, and configuration C differed from configuration B only by the addition of the corrugated liner.

#### Instrumentation

Engine-inlet air flow was determined by pressure and temperature measurements at the compressor inlet (station 1, fig. 6(a)). Instrumen-

2937 tation measured the engine midframe air bleed, which was subtracted from the engine-inlet air flow in order to obtain the afterburner air flow. Turbine-outlet temperature was calculated from engine performance, and turbine-outlet temperature for the electronic control was determined by averaging the engine manufacturer's eight thermocouples. The angle of whirl of the gas flow was measured at the diffuser outlet by means of a rotatable rake (fig. 6(b)). Twenty-five thermocouples attached to the afterburner shell measured temperatures at station 8, located  $79\frac{5}{8}$  inches downstream of the turbine-outlet flange (fig. 6(c)). Total pressures at the exhaust-nozzle inlet were obtained with a water-cooled survey rake (fig. 6(d)), and ambient pressure in the region of the exhaust-nozzle outlet was determined by static-pressure probes in the plane of the exhaust nozzle. Engine and afterburner fuel flows were measured by calibrated rotameters. The fuel used in this investigation was MIL-F-5624A grade JP-4.

#### PROCEDURE

The inlet and exhaust conditions for these tests were determined by the altitude and flight Mach number according to NACA standard atmosphere; 100-percent ram pressure recovery was assumed. Afterburner performance data were obtained over a range of altitudes from 15,000 to 50,000 feet at a flight Mach number of 0.6 and a range of flight Mach numbers from 0.4 to 1.0 at an altitude of 30,000 feet. For each flight condition, data were obtained at rated engine speed and turbine-outlet temperature as maintained by the electronic integral control for a range of fuel-air ratios.

The range of fuel-air ratios at each flight condition represents, in general, the practical operating range for the afterburner. The rich operating limit was determined by afterburner shell temperature limit ( $2010^{\circ}$  R) or by maximum nozzle opening. It should be noted that, with the electronic control, exhaust-nozzle area is a function of exhaust-gas temperature. Therefore, maximum nozzle opening was either the physical limit of the nozzle or the area obtained with maximum temperature for those configurations that reached a point of decreasing temperature with increasing fuel flow. The lean limit of operation was indicated by unsteady (oscillatory) combustion in the afterburner or blow-out of the flame. Over the range of conditions investigated (turbine-outlet pressures up to 2950 lb/sq ft with fuel-air ratios up to 0.04), this unsteady combustion at the lean limit was the only oscillatory combustion phenomenon encountered. No significant altitude limits were obtained for these configurations because of limitations of the altitude exhaust facilities. For afterburning conditions, the facility was limited to about 54,000 feet, which corresponds to a turbine-outlet total pressure of about 580 pounds per square foot.

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The jet thrust produced by the engine and afterburner was measured by a self-balancing null-type pneumatic thrust cell. A sketch of the linkage from the engine to the thrust cell is presented in figure 1(b). The symbols and methods of calculation are presented in the appendix.

## RESULTS AND DISCUSSION

The results of this investigation are discussed in relation to the three factors considered for improving the afterburner performance; that is, (1) reduction of whirl in the turbine-outlet gas, (2) matching of the fuel distribution with the mass-flow distribution, and (3) cooling of the afterburner shell. The altitude performance data are presented in tabular form in table II and in graphical form in figures 7 to 17.

### Effects of Turbine-Outlet Gas-Straightening Vanes and Vortex Generators on Afterburner Performance

The performance of an afterburning engine may be detrimentally affected by high angle of whirl of the turbine-outlet gases (ref. 6). The addition of a whirl velocity produces high resultant velocities that make burning more difficult. Furthermore, a high angle of whirl produces high total-pressure losses in the burner that are particularly noticeable during nonafterburning operation. Whirl losses are less pronounced during combustion, probably because the high level of turbulence during combustion reduces the whirl component of the flow.

Since whirl angle increases with diffusion (ref. 7), measurements of the whirl were made at the diffuser outlet to obtain the maximum angle and the results are presented in figure 7. The method of obtaining this curve is the same as used in reference 6. The direction of whirl was opposite to turbine rotation, as indicated by the negative angles on figure 7, and was greater than  $30^\circ$  over most of the passage. The installation of the turbine-outlet gas-straightening vanes and vortex generators, designed by the manufacturer to reduce the whirl angle and to provide a uniform velocity profile, resulted in only  $10^\circ$  whirl in the same counterrotational direction.

Effect on afterburner-inlet temperature and pressure profiles. - Installation of the turbine-outlet gas-straightening vanes and vortex generators caused a shift in afterburner-inlet temperature profile in relation to the control thermocouple. After the vanes and vortex generators were installed, the average burner-inlet gas temperature was from  $40^\circ$  to  $70^\circ$  R lower for the same indicated control temperature.

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This effect can be seen by inspection of data from table II. The setting of turbine-outlet temperature with the manufacturer's instrumentation resulted in operation at a lower turbine-outlet temperature and resulting lower nonafterburning thrusts for the configurations having straightening vanes and vortex generators.

Another effect of the straightening vanes and vortex generators was a slightly more uniform total-pressure distribution (fig. 8) at the diffuser-outlet (station 6). This trend in total-pressure profile indicates more nearly uniform mass-flow and velocity distribution of the combustion gases. However, a comparison of the pressure levels on figure 8 is invalid because of the different temperature levels.

Effect on internal performance. - A comparison of the total-pressure-loss ratios before and after the straightening vanes and vortex generators were installed is presented in figure 9. In the configuration with vanes and vortex generators (configuration B), the turbine-outlet total-pressure measurements were made downstream of the vanes. Therefore, the total-pressure loss through the straightening vanes and vortex generators was not included in the pressure-loss ratio, defined as the ratio of the total-pressure loss through the burner to the burner-inlet total pressure. The pressure-loss ratio through the vanes and vortex generators was determined to be approximately 0.025 at 30,000 feet and 0.035 at 50,000 feet from the engine pumping characteristics. This value has been added to the data on figure 9 for configuration B, and the combined pressure loss is shown by the dashed curve. Thus the installation of the straightening vanes and vortex generators reduced the over-all pressure-loss ratio by 0.02 to 0.01, the greater reduction occurring at low fuel-air ratios where the whirl is believed to have been greater.

The afterburner combustion efficiencies before (configuration A) and after (configuration B) straightening vanes and vortex generators were installed are compared in figure 10(a). For the range of fuel-air ratios investigated there was no appreciable difference in combustion efficiency at 30,000 feet, while at 50,000 feet the configuration with vanes and vortex generators had lower combustion efficiency. This loss in efficiency with vanes was due to a poor fuel-air distribution. The fuel distribution was designed for the mass-flow profile that existed before the vanes were installed. Following installation of the vanes, this mass-flow profile was modified, but the fuel distribution remained the same and resulted in a different fuel-air ratio profile.

The decrease in combustion efficiency for configuration B at 50,000 feet resulted in lower exhaust-gas temperatures (fig. 10(b)). The lower turbine-outlet temperatures also contributed to the decrease in afterburner exhaust-gas temperature.



Effect on over-all performance. - As previously noted, the effect of the straightening vanes and vortex generators was to reduce the afterburner pressure loss, and an increase in thrust would be expected. However, because of the difference in turbine-outlet conditions in this investigation, the increase in thrust was not realized and the augmented jet thrust ratio was the only valid basis for thrust comparison. (The nonafterburning jet thrust was calculated for each point with the turbine-outlet conditions obtained at that point and with the pressure drop assumed through a standard tail pipe as explained in the appendix.) The augmented jet thrust ratios are presented in figure 10(c). At 30,000 feet, where combustion efficiency was essentially the same, installation of the straightening vanes and vortex generators resulted in an increased jet thrust ratio of about 0.03. At 50,000 feet, however, the reduction in combustion efficiency countered the reduction in afterburner pressure loss, and an increase of only 0.01 was observed in jet thrust ratio.

Net thrust specific fuel consumption, presented in figure 10(d), is affected by the tail-pipe pressure loss, turbine-outlet gas temperature, and combustion efficiency. At 30,000 feet, combustion efficiency was unaffected by the turbine-outlet gas-straightening vanes, and the reduced tail-pipe pressure loss should have resulted in decreased specific fuel consumption. However, the lack of improvement illustrates the fact that the control should be rescheduled to give the same engine operating conditions after the temperature profile shift due to the installation of the straightening vanes. Also, the shift in mass-flow profile due to the addition of the vanes required an alteration to the fuel distribution to prevent a decrease in combustion efficiency at 50,000 feet. The results of modifying the fuel distribution after straightening vanes and vortex generators were installed are discussed in the following section.

#### Effect of Fuel Distribution on Performance

The installation of the turbine-outlet straightening vanes and vortex generators resulted in a loss in combustion efficiency that was attributed to a poor fuel-air distribution. Before alteration of the fuel system, a ceramic-coated corrugated liner was installed, the effects of which will be discussed later. Configurations C, D, and E differed only in fuel distribution.

A comparison of the combustion efficiency of configuration C, which had a fuel distribution equivalent to the original configuration

A, and configuration D, which had a fuel distribution designed to match the mass-flow profile, is presented in figure 11(a). The peak efficiency of configuration D occurred at higher fuel-air ratios than it did for configuration C. Also, configuration D had about 0.11 higher combustion efficiency than configuration C at 50,000 feet. For example, at 50,000 feet, combustion efficiency for configuration D had a peak value of 0.82 at a fuel-air ratio of 0.037 and was about 0.70 at a fuel-air ratio of 0.06.

The exhaust-gas temperature (fig. 11(b)) and net thrust ratio (fig. 11(c)) were higher for configuration D above a fuel-air ratio of 0.035 at 30,000 and 40,000 feet and higher at all fuel-air ratios investigated at 50,000 feet. Net thrust specific fuel consumption (fig. 11(d)) was lower for configuration D above a fuel-air ratio of 0.035 at 30,000 and 40,000 feet and lower at all fuel-air ratios investigated at 50,000 feet. The redesigned fuel distribution provided an improvement in combustion efficiency at fuel-air ratios in excess of 0.035 and, hence, in over-all afterburner performance.

Comparison of configurations D and E (figs. 11(a) to (c)) shows that the use of equal-diameter spray holes with various spacing (configuration E) resulted in about the same performance as with various-diameter spray holes with equal spacing (configuration D). Furthermore, the use of a metering orifice at the inlet to the spray bar did not affect performance.

#### Afterburner Shell Cooling

A maximum temperature limit of  $2010^{\circ}$  R for the afterburner shell was imposed by the structural strength requirements of the afterburner. The shell temperature of an afterburner is a function of the exhaust-gas temperature and the burner pressure (gas density). It was found that with a rich fuel-air mixture the original configuration A was limited at 30,000 feet and below by maximum shell temperature. Therefore, to permit higher exhaust-gas temperatures at low altitudes, methods of cooling the afterburner shell were investigated.

Reduction in turbine-outlet whirl. - A reduction in the temperature of the afterburner shell of from  $80^{\circ}$  to  $120^{\circ}$  R resulted from the installation of the straightening vanes and vortex generators (fig. 12) at an exhaust-gas temperature of  $3100^{\circ}$  R and altitudes of 30,000 and 50,000 feet. The larger reduction occurred at the lower altitude, where shell cooling was more critical. The data further indicate that the reduction would have been greater at higher fuel-air ratios. This decrease in shell temperature with the installation of straightening vanes was due to two factors. The decrease in whirl reduced the

tendency for the fuel to centrifuge toward the outer shell and also probably increased the thickness of the boundary layer at the outer shell.

Fuel distribution moved away from shell. - A previous investigation (ref. 5) showed that shell temperatures could be lowered by modifying the fuel distribution to reduce the amount of burning near the shell. Configuration F incorporates such a modification, as shown by a comparison of figures 3(a) and (e). This modification reduced the shell temperature approximately  $30^{\circ}$  at 30,000 feet (fig. 12). However, it lowered the altitude limit from over 54,000 feet for configuration A to 50,000 feet for configuration F at a flight Mach number of 0.6.

Ceramic-coated corrugated liner. - Configuration C incorporated a ceramic-coated corrugated liner installed in the afterburner in addition to the straightening vanes and vortex generators. Over the range of exhaust-gas temperatures investigated, the combination of straightening vanes and liner reduced the temperature  $200^{\circ}$  to  $300^{\circ}$  R. For configuration C an increase in exhaust-gas temperature from  $2800^{\circ}$  to  $3300^{\circ}$  R at 30,000 feet (fig. 12(b)) resulted in only  $50^{\circ}$  R increase in shell temperature. The combination of straightening vanes and liner was the most effective means of reducing shell temperatures investigated and was equivalent to the use of approximately 8 percent of engine air flow for cooling air in a conventional convective cooling system.

The coating used on the liner was a Bureau of Standards number A418 ceramic. After approximately 33 hours of afterburning, the coating and liner showed no evidence of deterioration.

A comparison of the shell temperatures for the three configurations (A, B, and C) over a range of turbine-outlet pressures with an exhaust-gas temperature of  $3100^{\circ}$  R is presented in figure 13. The temperature for the original configuration A increased with increasing pressure, while that for configuration C (liner plus straightening vanes and vortex generators) was essentially constant. This figure illustrates the greater effectiveness of the vanes and liner at low altitudes (high burner pressures), where shell temperatures were found to be critical on the original configuration.

#### Performance of Final Configuration

Performance data are presented for configuration D over a range of altitudes from 15,000 to 50,000 feet at a flight Mach number of 0.6, and for flight Mach numbers of 0.4 and 0.8 at an altitude of 30,000

feet. This configuration was used for evaluation because it had performance and operating and cooling characteristics equivalent or superior to any of the other configurations.

Effect of altitude and flight Mach number variations. - For a given afterburner fuel-air ratio, increasing the altitude at a constant flight Mach number tended to lower the combustion efficiency, exhaust-gas temperature, and augmented net thrust ratio (fig. 14). The peak combustion efficiency decreased from 96 percent for an afterburner fuel-air ratio of 0.027 at 15,000 feet to 82 percent for a fuel-air ratio of 0.037 at 50,000 feet. The corresponding exhaust-gas temperatures were 2960° and 2880° R. For a constant fuel-air ratio of 0.03, the temperature decreased from 2960° R at 15,000 feet to 2710° R at 50,000 feet, while the combustion efficiency decreased from 96 to 76 percent. For a wide-open exhaust nozzle, the exhaust-gas temperature decreased from 3550° R at 30,000 feet at a fuel-air ratio of 0.056 (high temperature data were not obtained at 15,000 ft) to 3380° R at 50,000 feet at a fuel-air ratio of 0.061. This decrease resulted in a decrease in augmented net thrust ratio from 1.6 to 1.52.

The net thrust specific fuel consumption decreased as the altitude increased up to an altitude between 30,000 and 40,000 feet. A further increase in altitude resulted in an increase in specific fuel consumption. This decrease was due to the increasing engine cycle efficiency with the decreasing compressor-inlet temperature as the altitude increased until the tropopause (35,000 ft) was reached, which more than compensated for the decrease in combustion efficiency. Above the tropopause, the temperature remained constant, and the decreasing combustion efficiency caused an increase in specific fuel consumption.

The effect of varying flight Mach number at an altitude of 30,000 feet is presented in figure 15. A decrease in flight Mach number affected the afterburner combustion efficiency (fig. 15(a)) and the exhaust-gas temperature (fig. 15(b)) in the same manner as did an increase in altitude; that is, combustion efficiency and exhaust-gas temperature decreased with decreasing Mach number for a given afterburner fuel-air ratio. However, since the range of flight Mach numbers was small (0.8 to 0.4), the effect was slight. The augmented net thrust ratio (fig. 15(c)) decreased with decreasing Mach number, but the net thrust specific fuel consumption (fig. 15(d)) increased as a result of the decreasing combustion efficiency.

Afterburner combustion efficiency as a function of turbine-outlet total pressure is presented in figure 16 for afterburner fuel-air ratios of 0.025, 0.035, 0.045, and 0.055. At the low fuel-air ratio, the efficiency decreased rapidly with decreasing pressure, but decreasing pressure affected the efficiency less at the higher fuel-air ratios.

### Thrust Generalization

A method of generalizing jet thrust for a turbojet engine has been developed in reference 8. The basis of this method is that the jet thrust is a function of the flow through a nozzle and may, therefore, be described in terms of the nozzle-inlet total pressure, the ambient static pressure, and the throat area of the nozzle. If the experimentally determined jet thrust is adjusted for nozzle area variation, it will generalize when plotted against  $(1.25P_g - p_0)$ . This method of generalization provides a calibration that may be used in determining turbojet engine jet thrust in a flight installation for take-off and flight conditions.

This method of generalizing jet thrust has been used for the data of configuration D (fig. 17). The factor used to adjust the jet thrust for area variations of the variable-area nozzle, which is presented in figure 17(b), is the ratio of jet thrust to jet thrust for maximum nozzle area as a function of the ratio of actual nozzle area to maximum nozzle area. Therefore, the jet thrust presented in figure 17(a) is the jet thrust obtainable with the maximum nozzle area.

The generalized thrust curve may be used to obtain the jet thrust for a given flight condition and afterburner fuel-air ratio. The values of the exhaust-nozzle-inlet total pressure and ambient static pressure may be obtained from the flight conditions and the engine characteristics. With these data, the jet thrust for maximum nozzle area can be obtained. The exhaust-nozzle area for a given afterburner fuel-air ratio can be obtained from figure 17(c), and from this the ratio of jet thrust at the given exhaust-nozzle area to the jet thrust at maximum nozzle area can be obtained. A numerical example of this calculation is given in the appendix.

### CONCLUDING REMARKS

An investigation of the J47-GE-17 afterburning engine showed that the installation of turbine-outlet gas-straightening vanes and vortex generators reduced the counterrotational whirl by approximately  $20^\circ$ . This change modified the burner-inlet (turbine-outlet) total-temperature profile so that the calculated average turbine-outlet gas temperature was from  $40^\circ$  to  $70^\circ$  R lower for the same indicated control temperature. This decrease in turbine-outlet temperature resulted in lower thrust for the same afterburner fuel-air ratio. However, the total-pressure-loss ratio was reduced by 0.01 to 0.02, and the augmented jet thrust ratio was increased by 0.01 to 0.03. Because of a change in total-pressure profile and thus a change in mass-flow profile with the same fuel distribution, the fuel-air distribution

at the flame holder was not as suitable with the vanes and vortex generators, and the combustion efficiency decreased.

By adjusting the fuel distribution to the mass-flow profile that resulted from the installation of the straightening vanes and vortex generators, the loss in combustion efficiency was overcome. The peak value at 50,000 feet altitude was approximately 82 percent at a fuel-air ratio of 0.037 and was about 70 percent at a fuel-air ratio of 0.06. The combustion efficiency, the exhaust-gas temperature, and the augmented net thrust ratio were higher at higher fuel-air ratios, while net thrust specific fuel consumption was lower for the modified fuel distribution. The final fuel distribution was achieved both with spray bars having equally spaced various-diameter holes and uniorifice spray bars having variously spaced equal-diameter holes.

Afterburner shell temperatures were reduced approximately  $100^{\circ}$  R by the installation of the straightening vanes and vortex generators for an exhaust-gas temperature of  $3100^{\circ}$  R at 30,000 feet altitude. An additional reduction of  $100^{\circ}$  R in shell temperature was achieved by the installation of a ceramic-coated corrugated liner. The liner was used for approximately 33 hours of afterburner operation with no noticeable deterioration or change in appearance.

Except at the lean limit of combustion, no oscillatory combustion was encountered during this investigation, which covered burner pressures up to 2950 pounds per square foot with fuel-air ratios up to 0.04.

Lewis Flight Propulsion Laboratory  
National Advisory Committee for Aeronautics  
Cleveland, Ohio, May 18, 1953



## APPENDIX - CALCULATIONS

## Symbols

The following symbols are used in this report:

A	cross-sectional area, sq ft
$C_V$	velocity coefficient, ratio of actual jet velocity to effective jet velocity
$F_j$	jet thrust, lb
$F_j'$	calculated nonafterburning jet thrust, lb
$F_n$	net thrust, lb
$F_n'$	calculated nonafterburning net thrust, lb
f	fuel-air ratio
g	acceleration due to gravity, 32.2 ft/sec <sup>2</sup>
h	enthalpy, Btu/lb
$h_c$	lower heating value of fuel, Btu/lb
M	Mach number
P	total pressure, lb/sq ft abs
$P_g'$	total pressure at exhaust-nozzle survey station in standard engine tail pipe, lb/sq ft abs
p	static pressure, lb/sq ft abs
R	gas constant, 53.4 ft-lb/(lb)(°R)
T	total temperature, °R
V	velocity, ft/sec
$V_{ef}$	effective velocity, ft/sec
$W_a$	air flow, lb/sec

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$W_{a,c}$  compressor leakage air flow, lb/sec  
 $W_f$  fuel flow, lb/hr  
 $W_f/F_n$  specific fuel consumption based on total fuel flow and net thrust, (lb/hr)/lb thrust  
 $W_g$  gas flow, lb/sec  
 $\gamma$  ratio of specific heats for gases  
 $\eta$  combustion efficiency

## Subscripts:

a air  
ab afterburner  
e engine  
f fuel  
g gas  
m maximum exhaust-nozzle area  
n exhaust-nozzle outlet, vena contracta  
nc nozzle cooling  
s labyrinth seal  
T total  
tc turbine cooling  
tp turbine pump  
0 free-stream conditions  
1 engine inlet  
3 compressor outlet at engine combustor inlet  
5 turbine outlet or tail-pipe diffuser inlet

- 9 exhaust-nozzle inlet  
10 exhaust-nozzle outlet

### Methods of Calculations

Flight Mach number and airspeed. - Flight Mach number and equivalent airspeed were calculated from engine-inlet total pressure and temperature and free-stream static pressure with complete total-pressure ram recovery assumed:

$$M_0 = \sqrt{\frac{2}{\gamma_1 - 1} \left[ \left( \frac{P_1}{P_0} \right)^{\frac{\gamma_1 - 1}{\gamma_1}} - 1 \right]}$$

and

$$V_0 = M_0 \sqrt{\gamma_1 g R T_1 \left( \frac{P_0}{P_1} \right)^{\frac{\gamma_1 - 1}{\gamma_1}}}$$

Air flow. - Air flow was determined from pressure and temperature measurements obtained in the engine-inlet annulus by the following equation:

$$W_{a,1} = 0.98 p_1 A_1 \sqrt{\frac{2 \gamma_1 g}{(\gamma_1 - 1) R T_1} \left[ \left( \frac{P_1}{P_1} \right)^{\frac{\gamma_1 - 1}{\gamma_1}} - 1 \right] \left( \frac{P_1}{P_1} \right)^{\frac{\gamma_1 - 1}{\gamma_1}}}$$

where the 0.98 accounts for the 0.02 leakage between measuring station and compressor inlet. Air flow at the compressor outlet (station 3) was obtained by deducting the compressor leakage, turbine and nozzle cooling-air flows, and compressor bleed air used to drive the turbine fuel pump:

$$W_{a,3} = W_{a,1} - W_{a,c} - W_{a,tc} - W_{a,nc} - W_{a,tp}$$

Gas flow. - Afterburner gas flow is

$$W_{g,9} = W_{a,3} + \frac{W_{f,e} + W_{f,ab}}{3600} + W_{a,tc}$$

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Fuel-air ratio. - The engine fuel-air ratio is given by the following equation:

$$f_e = \frac{W_{f,e}}{3600 W_{a,3}}$$

The afterburner fuel-air ratio used herein is defined as the weight flow of fuel injected into the afterburner plus the unburned engine fuel divided by the weight flow of unburned air entering the afterburner. By combining air flow, engine fuel flow, afterburner fuel flow, and engine combustion efficiency, the following equation for afterburner fuel-air ratio is obtained:

$$f_{ab} = \frac{(1 - \eta_e)W_{f,e} + W_{f,ab}}{3600(W_{a,3} + W_{a,tc}) - \frac{\eta_e W_{f,e}}{0.0675}}$$

where 0.0675 is the stoichiometric fuel-air ratio for the engine fuel.

The total fuel-air ratio for the engine and afterburner is

$$f = \frac{W_{f,e} + W_{f,ab}}{3600(W_{a,3} + W_{a,tc})}$$

Engine combustor efficiency. - Engine combustor efficiency is the ratio of the enthalpy rise through the engine divided by the product of engine fuel flow and the lower heating value of the fuel:

$$\eta_e = \frac{(h_{a,5} - h_{a,1}) + f_e \lambda_e}{f_e h_c}$$

where  $\lambda = \frac{Am + B}{m + 1}$  (symbols of ref. 9).

Engine combustion efficiency was obtained from a correlation of the engine with a standard tail pipe. For this investigation  $T_5$  was then obtained by solving the above equation for  $h_{a,5}$ .

Afterburner combustion efficiency. - Afterburner combustion efficiency was obtained by dividing the enthalpy rise through the afterburner by the product of the afterburner fuel flow and lower heating value of the fuel:

$$\eta_{ab} = \frac{(h_{a,10} - h_{a,1}) + f \lambda_{10} - \eta_e f_e h_c}{f h_c - \eta_e f_e h_c}$$

Augmented thrust. - The jet thrust of the combined engine and afterburner was determined from the thrust-system measurements by the equation

$$F_j = F_d + A_s(P_1 - p_0) + 0.80 \left( \frac{1}{2} \frac{W_{a,1}}{0.98g} V_0 \right)$$

where  $F_d$  is equal to the thrust-system scale reading adjusted for the pressure difference on the link connecting the thrust bed in the test chamber and the measuring cell outside the test chamber, and the last term  $0.80 \left( \frac{1}{2} \frac{W_{a,1}}{0.98g} V_0 \right)$  is the difference between the pressure forces on the bellmouth and the momentum at the bellmouth outlet.

The augmented net thrust was obtained by subtracting the free-stream momentum of the inlet air from the jet thrust:

$$F_n = F_j - \frac{W_{a,1}}{0.98g} V_0$$

where the  $\frac{W_{a,1}}{0.98}$  is the air flow into the bellmouth.

Nonafterburning jet thrust. - The jet thrust for the nonafterburning engine was calculated for each afterburning point with measured turbine-outlet total pressure, engine gas flow, calculated turbine-outlet total temperature, and an assumed tail-pipe pressure drop. Experimental data from the nonafterburning engine indicated that the total-pressure loss through a standard tail pipe between stations 5 and 9 was approximately 0.04 at rated engine speed, that is,  $P_9 = 0.96P_5$ . The nozzle velocity coefficient for the nonafterburning engine was assumed to be 0.99, while that for the afterburning engine was assumed to be 0.97. The jet thrust equation used was

$$F'_j = C_v W_{g,5} \sqrt{\frac{RT_5}{g}} \left( \frac{V_{ef}}{\sqrt{gRT}} \right)$$

where  $(V_{ef}/\sqrt{gRT})$  is an effective velocity parameter derived in reference 10; and the nonburning net thrust was

$$F'_n = F'_j - \frac{W_{a,1}}{0.98g} V_0$$

Exhaust-gas total temperature. - The total temperature of the exhaust gas was calculated from the jet thrust and conditions existing at the exhaust nozzle:

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$$T_{10} = \frac{1}{C_V^2} \frac{g}{R} \left[ \frac{F_j}{W_{g,9}} \left( \frac{1}{\sqrt{gRT}} \right) \right]^2$$

where  $\left( \frac{V_{ef}}{\sqrt{gRT}} \right)$  is the effective velocity parameter.

Calculated jet thrust from generalized jet thrust. - It has been shown (ref. 8) that the jet thrust of a turbojet engine may be related to the nozzle-inlet total pressure, ambient static pressure, and area. This relation is

$$F_j \propto A_{10}(1.25P_9 - p_0)$$

This equation may be represented by two plots as figures 17(a) and (b).

For an assumed flight condition of 40,000 feet altitude and a flight Mach number of 0.6, the pressure values are found from table II as  $P_9 = 1009$  pounds per square foot and  $p_0 = 392$  pounds per square foot. Then

$$1.25P_9 - p_0 = 1.25(1009) - 392 = 869.5 \text{ lb/sq ft}$$

From figure 17(a), the jet thrust for an open nozzle ( $A_{10,m} = 3.5$  sq ft), which would require an afterburner fuel-air ratio of 0.0575, is 2790 pounds.

If an afterburner fuel-air of 0.040 is desired,

$$A_{10} = 3.17 \text{ sq ft (from fig. 17(c))}$$

and

$$\frac{A_{10}}{A_{10,m}} = \frac{3.17}{3.50} = 0.906$$

With the area ratio, the thrust correction ratio may be found from figure 17(b) to be

$$F_j/F_{j,m} = 0.912$$

Therefore, the jet thrust for this fuel-air ratio is

$$F_j = F_{j,m} (F_j/F_{j,m}) = 2790(0.912) = 2544 \text{ lb}$$

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CZ-3 back

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TABLE I. - MODIFICATIONS INCORPORATED



Configuration	Number of fuel-spray bars	Orifices per spray bar	Mixing length, in.	Spray-bar location (fig. 2(a))	Fuel distribution (fig. 3)	Modifications
A	19	10	26	1	(a)	Manufacturer's original configuration
B	20	10	22	2	(b)	Fuel distribution equivalent to configuration A; turbine-outlet gas-straightening vanes and vortex generators added
C	20	10	22	2	(b)	Ceramic-coated corrugated liner added to configuration B
D	20	18	22	2	(c)	Modified fuel distribution; other variables same as configuration C; final configuration
E	20	16	22	2	(d)	Uniorifice spray bar with distribution of configuration D; other variables same as configuration C
F	19	10	26	1	(e)	Fuel distribution away from shell; no turbine-outlet straightening vanes, vortex generators, or coated liner

TABLE II. - ALTITUDE

Altitude, ft	Flight Mach number, M <sub>0</sub>	Free-stream static pressure, P <sub>0</sub> , lb sq ft abs	Engine- inlet total pressure, P <sub>1</sub> , lb sq ft abs	Engine- inlet total tempera- ture, T <sub>1</sub> , °R	Turbine- outlet total pressure, P <sub>5</sub> , lb sq ft abs	Turbine- outlet total tempera- ture, T <sub>5</sub> , °R	Manufac- turer's turbine- outlet control tempera- ture, T <sub>5,m</sub> , °R	Engine- inlet air flow, W <sub>a,i</sub> , lb sec	Engine fuel flow, W <sub>f,e</sub> , lb hr	Afterburner fuel flow, W <sub>f,ab</sub> , lb hr	Total fuel- air ratio, f	Afterburner fuel-air ratio, f <sub>ab</sub>
Original												
15,000	0.589 .600 .600 .581 .589	1210 1197 1198 1207 1199	1530 1526 1528 1528 1527	547 546 546 550 549	2978 2988 2981 2976 2961	1776 1783 1781 1772 1771	1755 1745 1760 1745 1750	72.50 72.12 72.76 72.24 72.08	4483 4463 4502 4426 4434	7040 5992 5367 4672 4011	0.0452 .0412 .0386 .0368 .0353	0.0374 .0321 .0286 .0248 .0216
30,000	0.367 .381 .383 .351 .383 .381	636 631 633 640 630 631	700 697 700 697 687 697	425 425 425 427 427 426	1614 1608 1611 1590 1606 1604	1803 1803 1803 1793 1798 1804	1770 1770 1770 1770 1765 1770	38.89 38.98 39.04 38.75 38.82 38.91	2690 2688 2688 2680 2672 2680	5303 4391 3629 2980 2408 1974	0.0583 .0217 .0480 .0413 .0250 .0340	0.0246 .0483 .0374 .0309 .0250 .0205
30,000	0.586 .575 .578 .585 .584 .584	632 640 635 636 628 633	797 801 796 800 798 797	441 441 441 433 440 441	1807 1803 1798 1823 1795 1799	1784 1779 1785 1778 1784 1779	1765 1760 1780 1760 1765 1760	43.60 43.76 43.51 44.00 43.85 43.38	2953 2954 2954 2953 2954 2937	4693 4311 3821 3380 2986 2330	0.0488 .0472 .0443 .0412 .0381 .0346	0.0430 .0392 .0351 .0308 .0268 .0215
30,000	0.783 .798 .792 .788 -0.802	630 633 630 634 637 630	972 959 958 959 959 962	471 468 467 466 466 466	2124 2074 2082 2080 2072 2093	1766 1771 1777 1769 1745 1769	1780 1760 1750 1750 1745 1745	51.15 50.61 50.71 50.79 50.67 50.84	3315 3297 3315 3315 3316 3320	5655 4874 4112 3351 2609 2194	0.0498 .0459 .0416 .0373 .0333 .0308	0.0435 .0379 .0319 .0280 .0204 .0171
30,000	0.985 .993 .993 .996 .995 .992	639 633 633 631 632 634	1188 1189 1188 1188 1188 1188	497 496 497 494 494 494	2524 2523 2556 2544 2646 2538	1789 1775 1807 1773 1775 1786	1780 1770 1795 1775 1775 1775	80.21 80.75 80.44 80.83 80.60 80.60	3948 3948 4013 3975 3985 3966	4571 4371 4052 3689 3304 2807	0.0402 .0389 .0379 .0357 .0342 .0316	0.0300 .0286 .0266 .0239 .0216 .0184
40,000	0.579 .581 .581 .570 .587 .570 .569	401 397 397 398 393 395 395	503 499 499 496 485 483 492	425 425 425 425 425 425 425	1149 1154 1154 1144 1150 1139 1143	1831 1831 1805 1831 1831 1844 1837	1780 1780 1775 1780 1780 1780 1780	27.77 27.67 28.10 27.53 27.47 27.36 27.38	1980 1952 1945 1945 1945 1962 1945	4473 3765 3284 2734 2312 1841 1638	0.0660 .0589 .0529 .0484 .0441 .0406 .0368	0.0653 .0553 .0470 .0404 .0343 .0292 .0239
50,000	0.643 .628 .643 .630 .608	239 242 239 239 243	315 316 316 312 312	436 433 430 419 421	710 710 711 724 720	1838 1822 1822 1818 1826	1770 1770 1765 1770 1765	16.97 17.13 17.24 17.38 17.30	1216 1221 1226 1240 1243	2807 2610 2261 1901 1672	0.0678 .0619 .0578 .0516 .0482	0.0679 .0622 .0536 .0457 .0401
Configuration B, straightening												
30,000	0.608 .605 .609 .605 .601	622 623 624 623 625	798 796 801 797 798	442 441 442 442 442	1727 1725 1730 1720 1720	1743 1741 1749 1743 1749	1765 1765 1760 1780 1780	43.64 43.77 43.69 43.57 43.62	2824 2824 2842 2824 2833	7040 6078 5240 4494 3362	0.0644 .0590 .0527 .0479 .0405	0.0633 .0544 .0472 .0406 .0304
40,000	0.611 .614 .612 .608	385 384 385 386	495 495 495 495	423 422 421 421	1087 1085 1085 1094	1755 1761 1753 1759	1770 1765 1770 1765	27.66 27.72 27.83 27.83	1840 1849 1849 1866	4113 3248 2348 1816	0.0614 .0524 .0430 .0376	0.0590 .0488 .0336 .0261
50,000	0.642 .643 .642 .653	235 235 235 235	310 310 310 313	425 424 425 424	665 665 666 673	1746 1768 1751 1754	1775 1775 1775 1775	17.21 17.19 17.24 17.29	1161 1176 1193 1171	2229 1820 1360 1075	0.0562 .0497 .0418 .0371	0.0518 .0428 .0321 .0255
Configuration C,												
15,000	0.611 .608 .610 .609 .610 .616 .610 .608	1186 1186 1188 1186 1183 1174 1162 1184	1525 1522 1526 1525 1520 1516 1519 1519	543 544 544 546 546 547 547 549	2890 2868 2898 2897 2855 2846 2856 2872	1778 1760 1753 1755 1765 1740 1744 1758	1770 1785 1765 1765 1765 1755 1750 1755	72.65 72.71 72.85 72.81 72.51 72.09 72.19 71.62	4492 4425 4415 4369 4369 4302 4311 4320	8200 7370 6520 5695 4815 3940 3115 2645	0.0499 .0462 .0428 .0395 .0361 .0326 .0293 .0277	0.0438 .0390 .0344 .0301 .0256 .0210 .0166 .0143

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## PERFORMANCE DATA

Exhaust- nozzle- inlet total pressure, $P_9$ , lb sq ft abs	Exhaust- nozzle- total temper- ature, $T_{10}$ , °R	Exhaust- nozzle area, $A_{10}$ , sq ft	Afterburner temperature ratio, $T_{10}/T_5$	Augmented net thrust, $F_J$ , lb	Augmented jet thrust ratio, $F_J/F_J$	Net thrust, $F_n$ , lb	Augmented net thrust ratio, $F_n/F_n$	Net thrust specific fuel con- sumption, $(W_{f,s}+W_{f,ab})/F_n$	Engine combustion efficiency, $\eta_s$	Afterburner combustion efficiency, $\eta_{ab}$	Maximum after- burner shell tempera- ture, $T_R$	Afterburner pressure- loss ratio, $P_5-P_9/P_5$
Configuration A												
2670	3131	3.04	1.763	6248	1.297	4778	1.427	2.412	0.990	0.863	1990	0.1034
2684	3000	2.92	1.663	6106	1.262	4617	1.379	2.264	.990	.916	1910	.1018
2679	2680	2.69	1.617	6013	1.235	4514	1.339	2.191	.990	.919	1871	.1013
2688	2755	2.62	1.543	5787	1.206	4314	1.297	2.109	.990	.907	1800	.0968
2677	2574	2.72	1.453	5591	1.167	4104	1.242	2.058	.990	.866	1735	.0969
1427	3362	3.22	1.865	3552	1.341	3110	1.409	2.570	0.990	0.765	2022	0.1159
1428	3274	3.10	1.816	3500	1.317	3041	1.384	2.328	.990	.847	1931	.1120
1455	3136	2.98	1.739	3417	1.284	2955	1.344	2.135	.990	.832	1805	.1093
1419	2982	2.92	1.663	3249	1.246	2826	1.292	1.992	.990	.831	1729	.1076
1444	2670	2.79	1.485	3123	1.181	2663	1.219	1.908	.990	.857	1682	.1009
1443	2538	2.69	1.407	3037	1.144	2578	1.174	1.805	.990	.874	1549	.1004
1610	3237	3.07	1.814	4130	1.326	3341	1.436	2.292	0.990	0.874	1934	0.1090
1606	3162	3.01	1.777	4057	1.307	3278	1.410	2.216	.990	.869	1830	.1093
1609	3030	2.99	1.697	3948	1.276	3169	1.369	2.138	.990	.865	1792	.1057
1638	2699	2.92	1.627	3918	1.248	3132	1.331	2.035	.990	.896	1800	.1031
1613	2761	2.82	1.548	3772	1.213	2974	1.287	1.954	.990	.897	1750	.1014
1622	2544	2.69	1.430	3567	1.163	2804	1.219	1.878	.990	.847	1663	.0984
1894	3245	3.10	1.837	5167	1.345	3896	1.519	2.302	0.990	0.862	2101	0.1083
1856	3089	3.01	1.744	4935	1.305	3690	1.454	2.214	.990	.868	1892	.1061
1870	2923	2.95	1.645	4601	1.260	3557	1.391	2.088	.990	.864	1776	.1018
1873	2701	2.82	1.564	4601	1.212	3354	1.351	1.968	.990	.867	1714	.0991
1872	2430	2.69	1.402	4370	1.156	3134	1.232	1.890	.990	.831	1681	.0981
1900	2368	2.56	1.339	4311	1.130	3050	1.194	1.808	.990	.822	1640	.0922
2277	2758	2.82	1.653	6097	1.273	4254	1.444	2.003	0.990	0.945	1926	0.0979
2273	2683	2.79	1.613	6050	1.255	4179	1.417	1.991	.990	.921	1888	.0991
2316	2637	2.75	1.570	6016	1.239	4155	1.387	1.941	.990	.936	1897	.0935
2302	2706	2.69	1.521	5889	1.215	4016	1.351	1.905	.990	.920	1817	.0911
2307	2639	2.65	1.472	5791	1.195	3927	1.316	1.856	.990	.926	1736	.0939
2306	2460	2.56	1.377	5569	1.153	3709	1.250	1.826	.990	.854	1720	.0914
1001	3472	3.35	1.896	2738	1.363	2249	1.480	2.860	0.987	0.639	1995	0.1268
1014	3306	3.22	1.806	2669	1.327	2180	1.431	2.622	.987	.726	1910	.1213
1018	3183	3.10	1.752	2644	1.302	2148	1.399	2.434	.987	.760	1870	.1170
1017	3101	3.04	1.694	2549	1.275	2068	1.361	2.267	.987	.820	1792	.1110
1016	3018	2.95	1.648	2514	1.252	2021	1.333	2.108	.987	.887	1745	.1148
1015	2861	2.85	1.552	2408	1.211	1934	1.277	2.013	.987	.867	1678	.1089
1021	2600	2.72	1.415	2298	1.154	1824	1.202	1.910	.987	.800	1598	.1067
612	3496	3.35	1.902	1897	1.364	1364	1.497	2.949	0.967	0.684	1817	0.1377
612	3411	3.31	1.872	1880	1.347	1352	1.472	2.834	.967	.727	1820	.1381
613	3383	3.22	1.857	1694	1.338	1348	1.480	2.585	.968	.760	1761	.1361
632	3144	3.10	1.729	1644	1.287	1316	1.387	2.387	.968	.776	1717	.1270
631	3104	3.04	1.700	1607	1.271	1289	1.363	2.261	.968	.829	1668	.1244
vanes and vortex generators added												
1563	3380	3.28	1.928	4275	1.411	3456	1.563	2.854	0.990	0.679	1769	0.0776
1601	3270	3.18	1.878	4203	1.385	3366	1.527	2.629	.990	.726	1719	.0719
1611	3220	3.12	1.841	4152	1.368	3330	1.503	2.427	.990	.795	1718	.0588
1606	3152	3.08	1.808	4073	1.348	3269	1.477	2.245	.990	.866	1702	.0663
1616	2880	2.95	1.647	3867	1.279	3056	1.361	2.027	.990	.904	1619	.0605
1002	3210	3.25	1.829	2656	1.368	2146	1.500	2.774	0.987	0.654	1681	0.0782
1005	3200	3.17	1.817	2638	1.354	2125	1.461	2.399	.987	.786	1680	.0737
1013	2968	3.03	1.695	2530	1.235	2017	1.403	2.081	.987	.894	1659	.0664
1026	2669	2.87	1.517	2395	1.223	1885	1.301	1.948	.986	.848	1629	.0622
607	3108	3.24	1.780	1611	1.336	1278	1.464	2.653	0.964	0.681	1600	0.0868
611	2988	3.12	1.692	1572	1.297	1239	1.410	2.418	.966	.730	1600	.0804
615	2613	2.97	1.492	1486	1.211	1152	1.290	2.225	.963	.665	1574	.0754
627	2571	2.82	1.466	1460	1.197	1120	1.273	2.005	.966	.776	1535	.0692
straightening vanes and liner												
2691	3228	3.12	1.816	6454	1.350	4936	1.512	2.571	0.990	0.825	1561	0.0889
2695	3106	3.06	1.765	6324	1.327	4810	1.480	2.452	.990	.843	1533	.0868
2711	3008	2.97	1.716	6226	1.306	4705	1.449	2.324	.990	.873	1550	.0845
2717	2887	2.91	1.634	6065	1.271	4541	1.399	2.220	.990	.878	1540	.0821
2674	2735	2.82	1.556	5831	1.235	4313	1.348	2.129	.990	.896	1519	.0834
2677	2670	2.70	1.474	5628	1.200	4102	1.296	2.009	.990	.902	1516	.0894
2699	2322	2.58	1.331	5338	1.138	3826	1.204	1.941	.990	.790	1494	.0560
2722	2153	2.49	1.225	5107	1.091	3609	1.133	1.930	.990	.633	1490	.0622

TABLE II. - Concluded.

Altitude, ft	Flight Mach number, $M_0$	Free-stream static pressure, $P_0$ , lb sq ft abs	Engine- inlet total pressure, $P_1$ , lb sq ft abs	Engine- inlet total temperature, $T_1$ , °R	Turbine- inlet total pressure, $P_2$ , lb sq ft abs	Turbine- outlet total temperature, $T_3$ , °R	Manufac- turer's turbine- outlet control tempera- ture, $T_{5,M}$ , °R	Engine- inlet air flow, $W_{a,1}$ , lb sec	Engine fuel flow, $W_{f,e}$ , lb hr	Afterburner fuel flow, $W_{f,ab}$ , lb hr	Total fuel- air ratio, $f$	Afterburner fuel-air ratio, $f_{ab}$
Configuration C,												
30,000	0.604 .598 .604 .609 .611 .616	.625 398 .624 802 620 .622	799 494 798 802 798 802	444 443 442 441 439 445	1725 1711 1721 1721 1724 1703	1751 1750 1749 1742 1746 1739	1765 1765 1765 1765 1765 1765	43.62 43.49 43.69 43.81 43.73 43.74	2833 2835 2833 2833 2850 2807	7775 8427 5137 3957 2879 1797	0.0693 .0807 .0520 .0442 .0373 .0300	0.0699 .0582 .0483 .0355 .0260 .0163
40,000	0.600 .597 .606 .619 .612	398 390 387 384 387	494 497 486 497 498	422 422 428 424 421	1093 1084 1091 1084 1093 1104	1768 1768 1770 1767 1765 1767	1765 1765 1765 1765 1765 1765	27.79 27.77 27.61 27.60 27.71 27.97	1865 1865 1849 1832 1857 1872	4431 3784 3006 2313 1607 1088	0.0644 .0580 .0500 .0430 .0358 .0300	0.0632 .0543 .0433 .0336 .0233 .0156
50,000	0.605 .596 .611 .606 .629	239 398 236 241 234	306 494 304 308 305	425 424 427 423 423	672 659 653 668 669	1801 1782 1758 1806 1806	1765 1765 1765 1765 1760	17.05 17.07 16.74 17.08 17.07	1197 1192 1142 1207 1202	2847 2227 1790 1297 669	0.0644 .0572 .0500 .0419 .0313	0.0631 .0531 .0433 .0314 .0167
Configuration D, modified fuel distribution												
15,000	0.614 .613 .614	1185 1188 1187	1527 1532 1531	539 533 537	2897 2947 2920	1737 1738 1741	1760 1780 1780	73.39 74.14 73.67	4387 4464 4434	8021 5800 4612	0.0482 .0395 .0350	0.0418 .0401 .0241
30,000	0.598 .594 .404 .412	634 532 826 626	707 704 700 704	427 426 425 423	1546 1536 1528 1554	1749 1742 1748 1740	1770 1760 1760 1755	39.35 39.24 39.05 39.39	2580 2542 2563 2580	5614 4193 3035 1980	0.0594 .0481 .0409 .0350	0.0553 .0422 .0308 .0200
30,000	0.610 .608 .610 .602 .603	624 628 624 625 630	802 805 802 798 805	438 436 444 445 445	1740 1734 1709 1712 1723	1748 1738 1745 1752 1746	1765 1760 1755 1755 1755	44.18 44.39 43.65 43.50 43.80	2876 2876 2816 2633 2835	6317 5281 4213 3025 2028	0.0593 .0522 .0459 .0384 .0334	0.0583 .0486 .0379 .0275 .0207
30,000	0.809 .814 .804 .800 .819	625 825 626 632 620	961 986 959 962 962	463 460 472 456 455	2027 2031 1995 2039 2042	1754 1739 1749 1742 1748	1770 1760 1765 1760 1760	51.21 51.75 50.48 51.66 51.80	3287 3297 3208 3297 3324	7706 5908 4795 3765 3007	0.0612 .0507 .0452 .0390 .0348	0.0589 .0446 .0372 .0286 .0229
40,000	0.600 .596 .594 .600 .592	391 393 394 394 393	498 498 499 502 498	415 416 423 423 422	1104 1103 1096 1097 1098	1755 1749 1749 1755 1761	1765 1765 1765 1765 1770	28.23 28.30 28.07 28.04 28.00	1888 1880 1858 1856 1865	4094 3814 2844 2128 1502	0.0604 .0543 .0477 .0406 .0343	0.0577 .0493 .0402 .0305 .0215
50,000	0.658 .636 .619 .653 .638 .637	255 237 239 238 234 235	315 311 310 312 308 309	420 419 418 417 417 418	679 678 678 672 668	1752 1770 1776 1747 1768 1769	1775 1772 1765 1770 1765 1775	17.55 17.34 17.25 17.46 17.32 17.33	1190 1190 1190 1182 1190 1190	2663 2905 2194 1874 1513 1229	0.0627 .0609 .0560 .0499 .0448 .0398	0.0609 .0583 .0516 .0433 .0356 .0290
Configuration E, modified fuel												
40,000	0.589 .610 .601 .667 .575 .584	396 389 394 403 401 397	500 500 503 501 501 500	421 423 423 423 422 423	1092 1085 1087 1092 1091 1080	1748 1742 1742 1742 1741 1742	1750 1745 1745 1745 1745 1745	28.07 28.19 28.15 28.12 28.24 28.15	1825 1817 1825 1825 1825 1826	3948 3680 3364 3188 2778 2285	0.0588 .0557 .0525 .0508 .0465 .0415	0.0553 .0514 .0470 .0346 .0387 .0320
50,000	0.632 .623 .626 .624	237 243 243 236	310 316 307 307	432 431 433 432	659 658 680 661	1793 1760 1773 1784	1770 1760 1770 1765	17.08 17.37 16.90 16.96	1171 1171 1142 1157	2717 2275 2050 1468	0.0649 .0586 .0535 .0441	0.0638 .0526 .0482 .0352
Configuration F, original with												
30,000	0.592 .590 .591 .592 .590 .590	634 631 632 631 633 636	804 798 800 796 801 804	443 441 440 445 440 438	1765 1759 1770 1778 1774 1779	1765 1774 1760 1765 1755 1778	1760 1755 1760 1755 1755 1755	43.67 43.44 43.67 43.23 43.74 43.92	2876 2876 2911 2868 2911 2928	6495 5855 4955 4311 3228 2243	0.0811 .0580 .0513 .0473 .0400 .0335	0.0587 .0516 .0451 .0396 .0294 .0204
40,000	0.590 .593 .584 .592 .588 .590	395 394 394 391 391 395	500 500 496 496 495 500	417 415 414 426 429 414	1149 1149 1149 1139 1130 1151	1812 1810 1792 1801 1816 1803	1780 1780 1780 1780 1775 1775	28.07 28.06 28.01 27.57 27.37 28.10	1980 1960 1936 1898 1903 1968	4073 3478 2934 2408 1901 1465	0.0813 .0552 .0486 .0455 .0398 .0346	0.0587 .0502 .0423 .0353 .0282 .0211

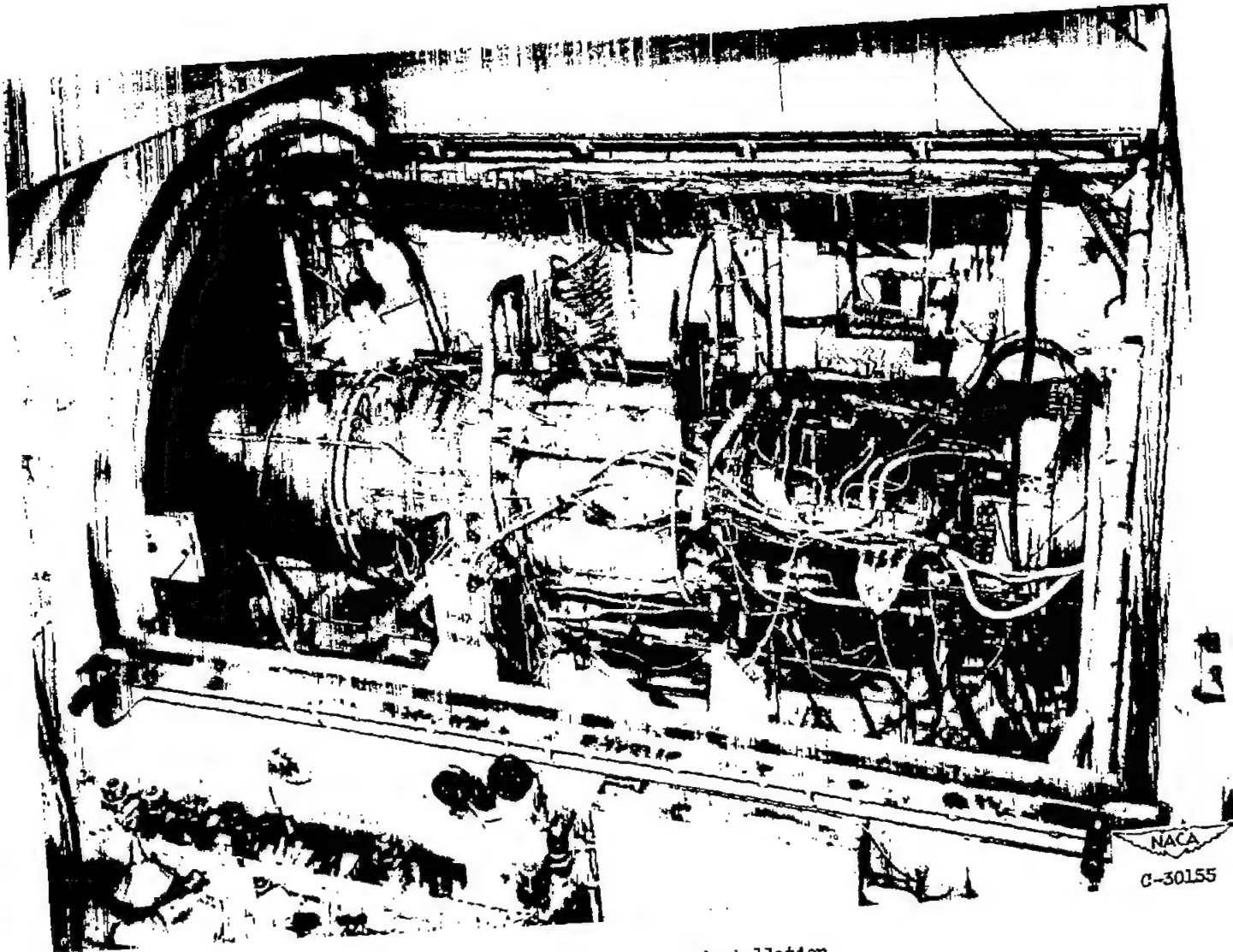
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## ALTITUDE PERFORMANCE DATA

Exhaust- nozzle- inlet total pressure, $P_9$ , lb sq ft abs	Exhaust- nozzle- total tempera- ture, $T_{10}$ , °R	Exhaust- nozzle area, $A_{10}$ , sq ft	Afterburner temperature ratio, $T_{10}/T_5$	Augmented jet thrust, $F_j$ , lb	Augmented jet thrust ratio, $F_j/F_j^*$	Net thrust, $F_n$ , lb	Augmented net thrust ratio, $F_n/F_n^*$	Net thrust specific fuel con- sumption, $(W_{f,ab} + W_{f,n})/F_n$	Engine combustion efficiency, $\eta_e$	Afterburner combustion efficiency, $\eta_{ab}$	Maximum- after- burner shell tempera- ture, °R	Afterburner pressure- loss ratio, $P_5 - P_9$ $P_5$
straightening vanes and linear												
1479	3271	3.31	1.888	4209	1.390	3393	1.535	3.126	0.990	0.589	1584	0.0836
1571	3508	3.29	1.890	4185	1.388	3370	1.531	2.748	.990	.704	1599	.0818
1584	3199	3.19	1.829	4103	1.354	3288	1.484	2.424	.990	.788	1588	.0796
1597	3048	3.07	1.750	3396	1.318	3173	1.436	2.140	.990	.904	1559	.0721
1613	2783	2.89	1.594	3809	1.252	2987	1.345	1.918	.990	.955	1540	.0644
1610	2329	2.58	1.339	3446	1.143	2612	1.198	1.763	.990	.844	1489	.0546
997	3170	3.30	1.793	2854	1.353	2150	1.478	2.928	0.987	0.600	1590	0.0882
1001	3186	3.24	1.802	2837	1.349	2135	1.469	2.648	.987	.689	1592	.0850
1000	3107	3.19	1.759	2584	1.322	2084	1.434	2.352	.987	.793	1610	.0836
1000	2978	3.03	1.681	2484	1.286	1975	1.389	2.089	.986	.898	1600	.0779
1019	2652	2.82	1.488	2372	1.210	1853	1.286	1.869	.987	.920	1491	.0677
1038	2238	2.62	1.550	2193	1.108	1677	1.146	1.753	.986	.727	1480	.0598
608	3026	3.31	1.680	1573	1.301	1260	1.408	3.049	0.987	0.535	1599	0.0952
596	3022	3.26	1.686	1554	1.294	1245	1.396	2.746	.986	.617	1600	.0966
596	2925	3.15	1.664	1491	1.280	1181	1.381	2.483	.983	.687	1610	.0913
615	2668	2.96	1.487	1452	1.202	1139	1.273	2.198	.987	.707	1499	.0816
620	2129	2.65	1.178	1297	1.062	973	1.085	1.923	.987	.486	1481	.0733
with straightening vanes and linear												
2662	3408	3.19	1.982	6662	1.392	5127	1.577	2.420	0.990	0.974	1612	0.0811
2750	2957	2.95	1.701	6301	1.294	4761	1.431	2.156	.990	.954	1582	.0669
2735	2728	2.82	1.567	5986	1.241	4446	1.353	2.035	.990	.948	1569	.0634
1409	3557	3.50	2.034	3669	1.424	3185	1.522	2.572	0.990	0.641	1819	0.0886
1414	3238	3.22	1.859	3470	1.355	2991	1.437	2.259	.990	.893	1570	.0794
1421	2916	3.01	1.668	3273	1.279	2785	1.345	2.010	.990	.927	1535	.0700
1464	2365	2.69	1.359	2994	1.153	2494	1.189	1.828	.990	.741	1489	.0579
1591	3548	3.50	2.032	4415	1.434	3587	1.594	2.563	0.990	0.838	1739	0.0856
1594	3326	3.31	1.831	4274	1.391	3447	1.535	2.361	.990	.882	1597	.0807
1585	3142	3.13	1.801	4036	1.340	3212	1.487	2.188	.990	.914	1560	.0728
1602	2828	2.92	1.614	3799	1.261	2987	1.357	1.961	.990	.943	1519	.0643
1625	2456	2.72	1.407	3553	1.174	2734	1.239	1.778	.990	.908	1490	.0569
1860	3586	3.50	2.033	5474	1.447	4198	1.675	2.619	0.990	0.807	1772	0.0824
1875	3346	3.22	1.824	5312	1.395	4019	1.599	2.290	.990	.904	1590	.0768
1884	3126	3.10	1.788	4953	1.343	3689	1.522	2.169	.990	.914	1549	.0707
1910	2870	2.92	1.648	4680	1.278	3598	1.419	1.964	.990	.947	1529	.0633
1921	2614	2.75	1.495	4669	1.216	3376	1.325	1.875	.990	.901	1500	.0583
1003	3407	3.50	1.941	2773	1.400	2266	1.537	2.640	0.987	0.760	1736	0.0915
1007	3306	3.29	1.890	2722	1.375	2216	1.503	2.434	.987	.817	1590	.0870
1006	3080	3.18	1.761	2583	1.321	2078	1.433	2.282	.987	.833	1580	.0822
1015	2835	2.98	1.615	2461	1.258	1952	1.348	2.041	.987	.878	1528	.0748
1027	2393	2.75	1.359	2255	1.151	1754	1.202	1.920	.987	.709	1493	.0647
611	3364	3.46	1.932	1729	1.394	1383	1.547	2.786	0.967	0.714	1609	0.1002
612	3339	3.37	1.886	1690	1.376	1359	1.516	2.719	.967	.717	1600	.0978
614	3229	3.28	1.818	1641	1.346	1320	1.470	2.564	.967	.743	1593	.0930
613	3132	3.16	1.793	1638	1.333	1298	1.460	2.354	.985	.814	1571	.0884
618	2814	2.98	1.592	1635	1.280	1205	1.357	2.243	.967	.738	1543	.0811
618	2694	2.89	1.523	1493	1.220	1162	1.301	2.082	.967	.788	1513	.0745
distribution with microrifice spray bar												
992	3465	3.50	1.982	2749	1.411	2250	1.563	2.566	0.985	0.815	----	0.0916
983	3308	3.22	1.899	2690	1.376	2173	1.511	2.330	.985	.791	----	.0940
988	3257	3.16	1.870	2657	1.362	2146	1.490	2.418	.985	.828	----	.0911
994	3218	3.10	1.847	2614	1.352	2130	1.469	2.333	.985	.843	----	.0898
998	3092	3.01	1.778	2568	1.322	2078	1.430	2.215	.985	.875	----	.0853
994	2907	2.89	1.669	2480	1.277	1982	1.373	2.074	.985	.901	----	.0796
598	3348	3.50	1.867	1657	1.375	1329	1.515	2.926	0.963	0.666	----	0.0956
606	3228	3.17	1.854	1622	1.355	1294	1.489	2.646	.963	.733	----	.0901
601	3146	3.06	1.774	1578	1.329	1255	1.453	2.527	.962	.743	----	.0894
607	2912	2.89	1.632	1514	1.266	1191	1.364	2.204	.961	.815	----	.0817
fuel distribution away from shell												
1564	3362	3.28	1.907	4189	1.856	3389	1.502	2.765	0.990	0.720	1912	0.1139
1561	3387	3.28	1.909	4166	1.844	3374	1.494	2.528	.990	.816	1949	.1126
1576	3231	3.16	1.815	4090	1.791	3286	1.442	2.389	.990	.829	1910	.1096
1573	3206	3.04	1.803	3976	1.765	3182	1.412	2.256	.990	.913	1790	.1047
1593	2834	2.92	1.600	3799	1.664	3004	1.316	2.044	.990	.886	1722	.1020
1608	2522	2.72	1.418	3581	1.560	2783	1.212	1.858	.990	.878	1600	.0961
1008	3323	3.25	1.854	2719	1.339	2221	1.450	2.716	0.987	0.701	1838	0.1227
1014	3280	3.19	1.812	2692	1.327	2193	1.433	2.478	.987	.781	1804	.1176
1002	3207	3.17	1.785	2631	1.306	2141	1.405	2.275	.987	.863	1832	.1211
1012	2970	2.98	1.649	2491	1.255	1995	1.340	2.157	.987	.842	1716	.1099
1010	2779	2.89	1.530	2393	1.205	1894	1.273	2.008	.987	.856	1661	.1082
1031	2418	2.75	1.341	2294	1.123	1798	1.163	1.909	.987	.716	1610	.1043

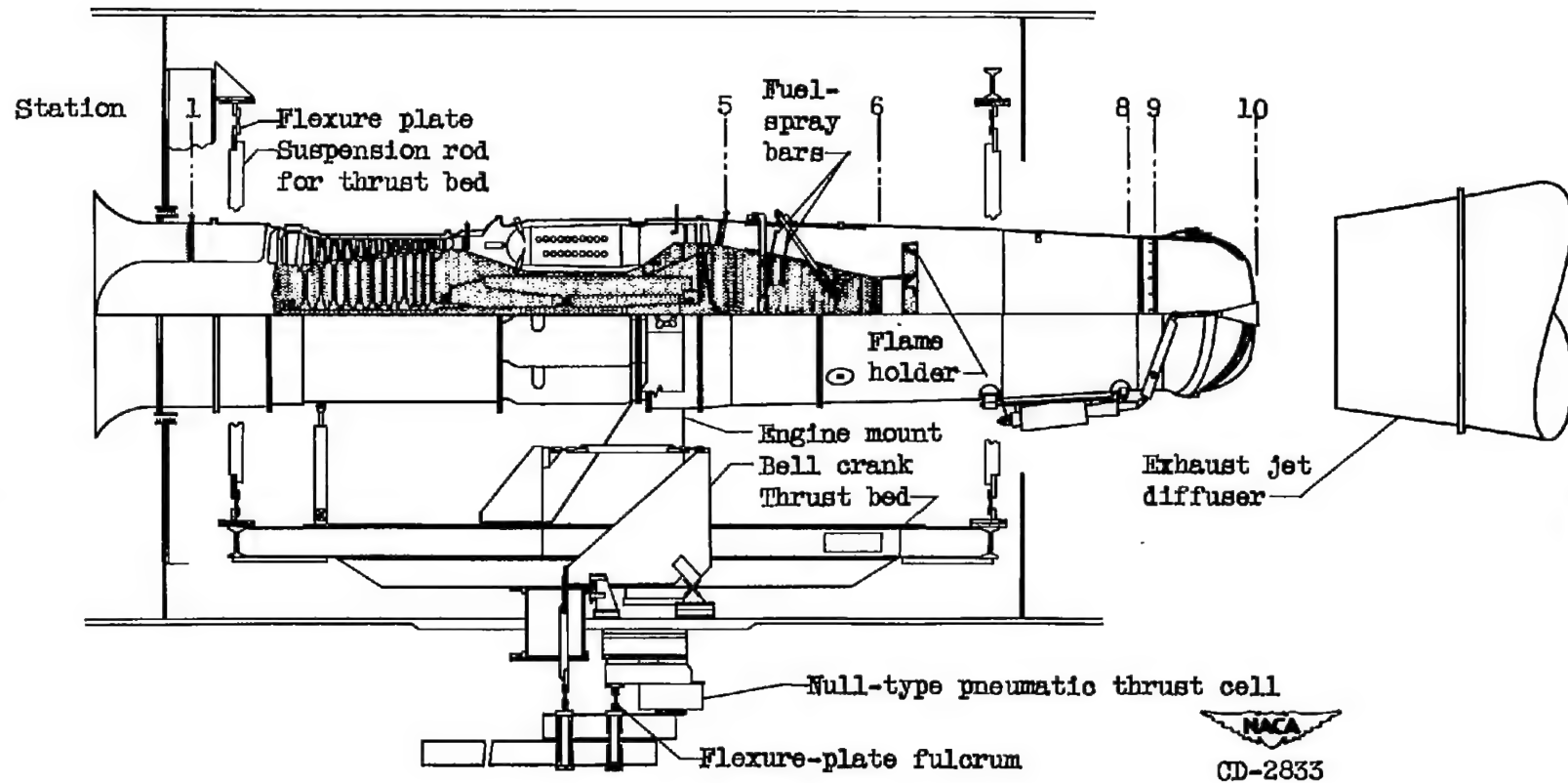
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(a) Photograph of installation.

Figure 1. - Turbojet engine equipped with afterburner and installed in altitude chamber.



(b) Schematic diagram of engine installation showing thrust-measuring system and instrumentation stations.

Figure 1. - Concluded. Turbojet engine equipped with afterburner and installed in altitude chamber.

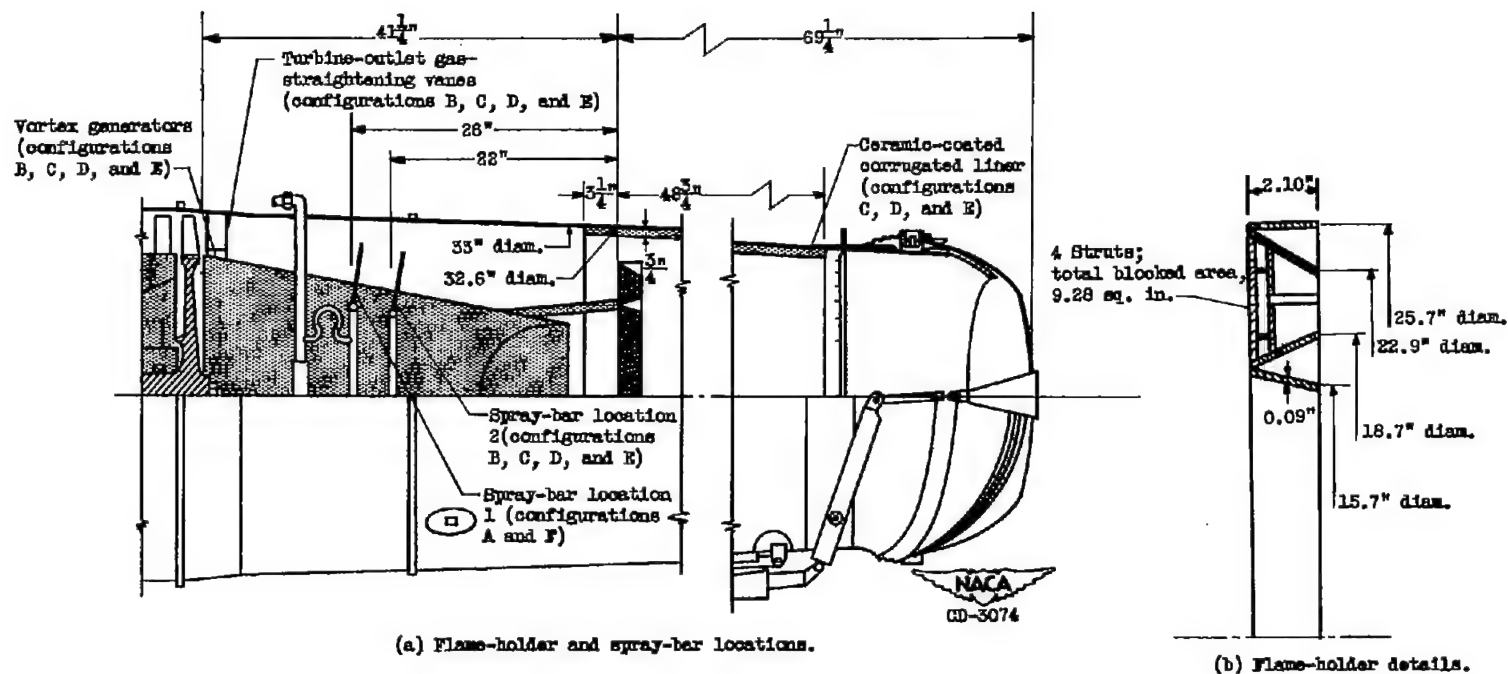
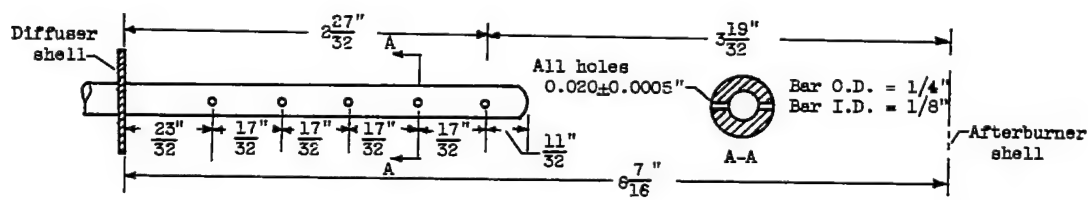
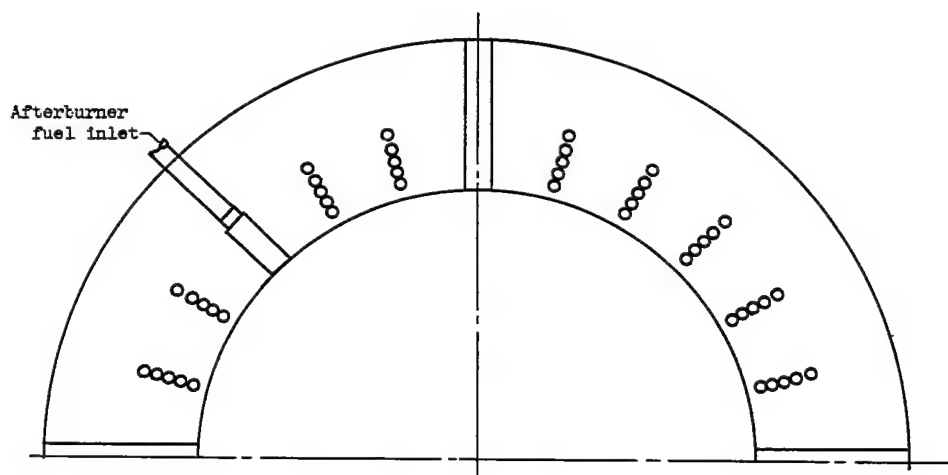
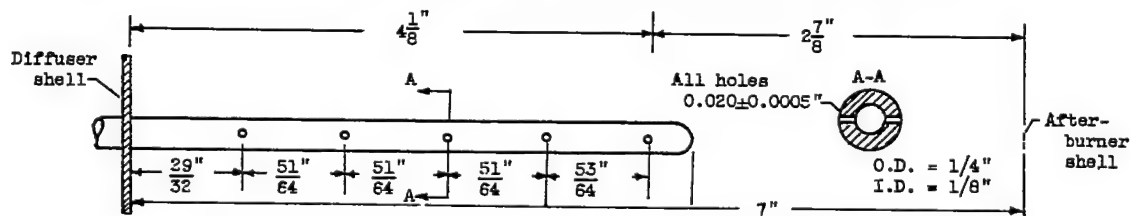
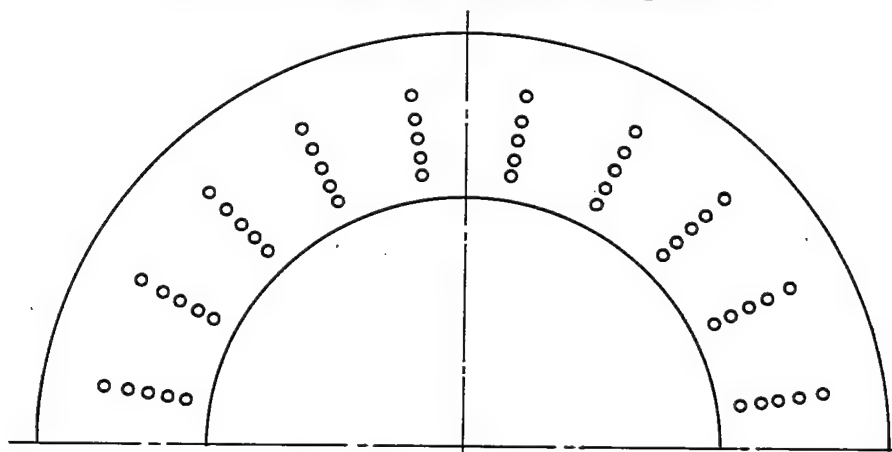


Figure 2. - Diagram of afterburner showing location of spray bars and flame holder.

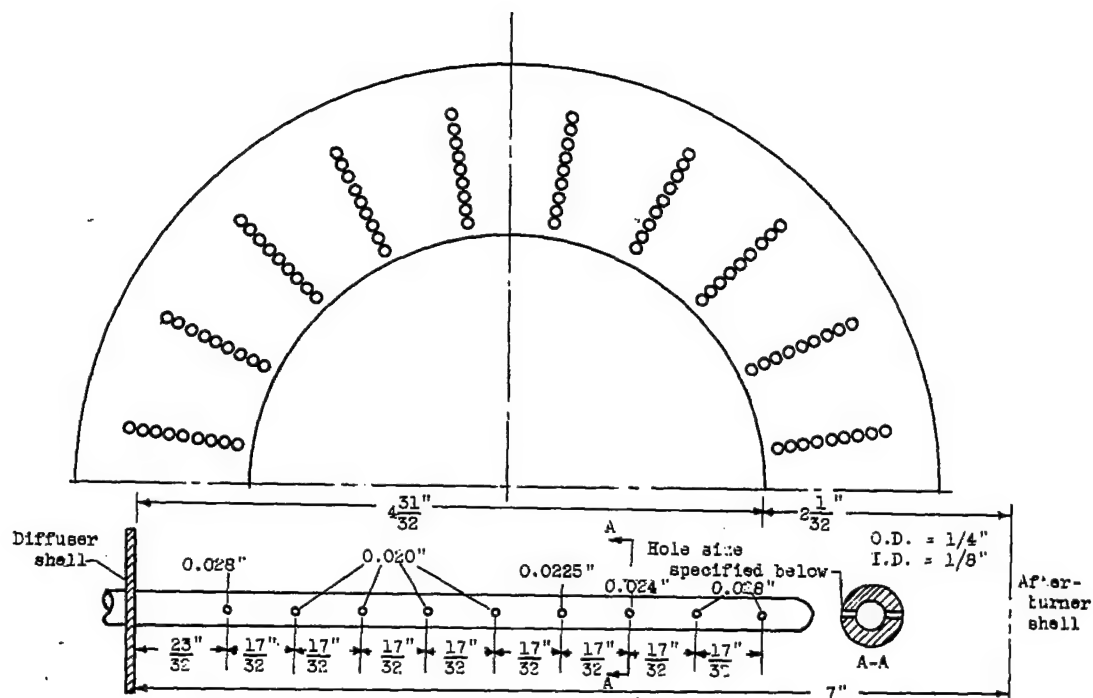


(a) Spray pattern and spray bar used in configuration A.

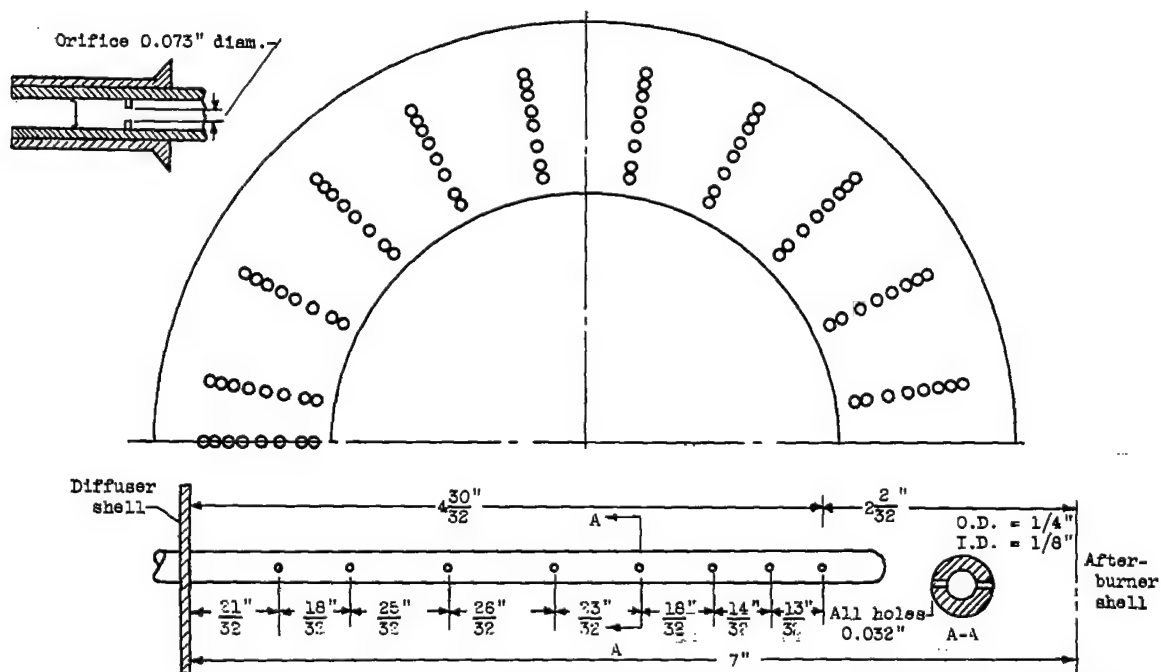


(b) Spray pattern and spray bar used in configurations B and C.

Figure 3. - Fuel-distribution patterns and spray-bar designs.



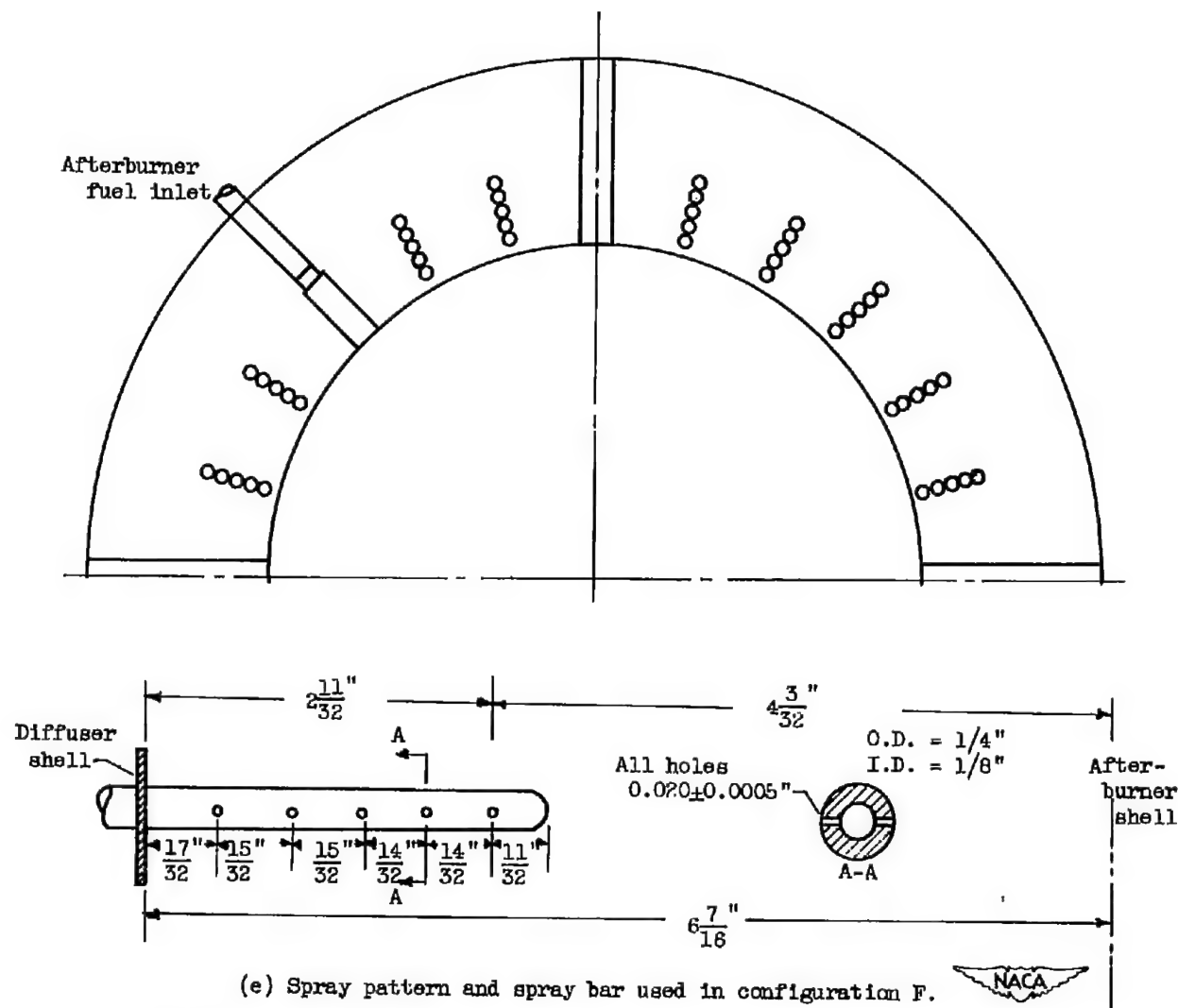
(c) Spray pattern and spray bar used in configuration D.



(d) Spray pattern and spray bar used in configuration E.

Figure 3. - Continued. Fuel-distribution patterns and spray-bar designs.

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(e) Spray pattern and spray bar used in configuration F.

Figure 3. - Concluded. Fuel-distribution patterns and spray-bar designs.



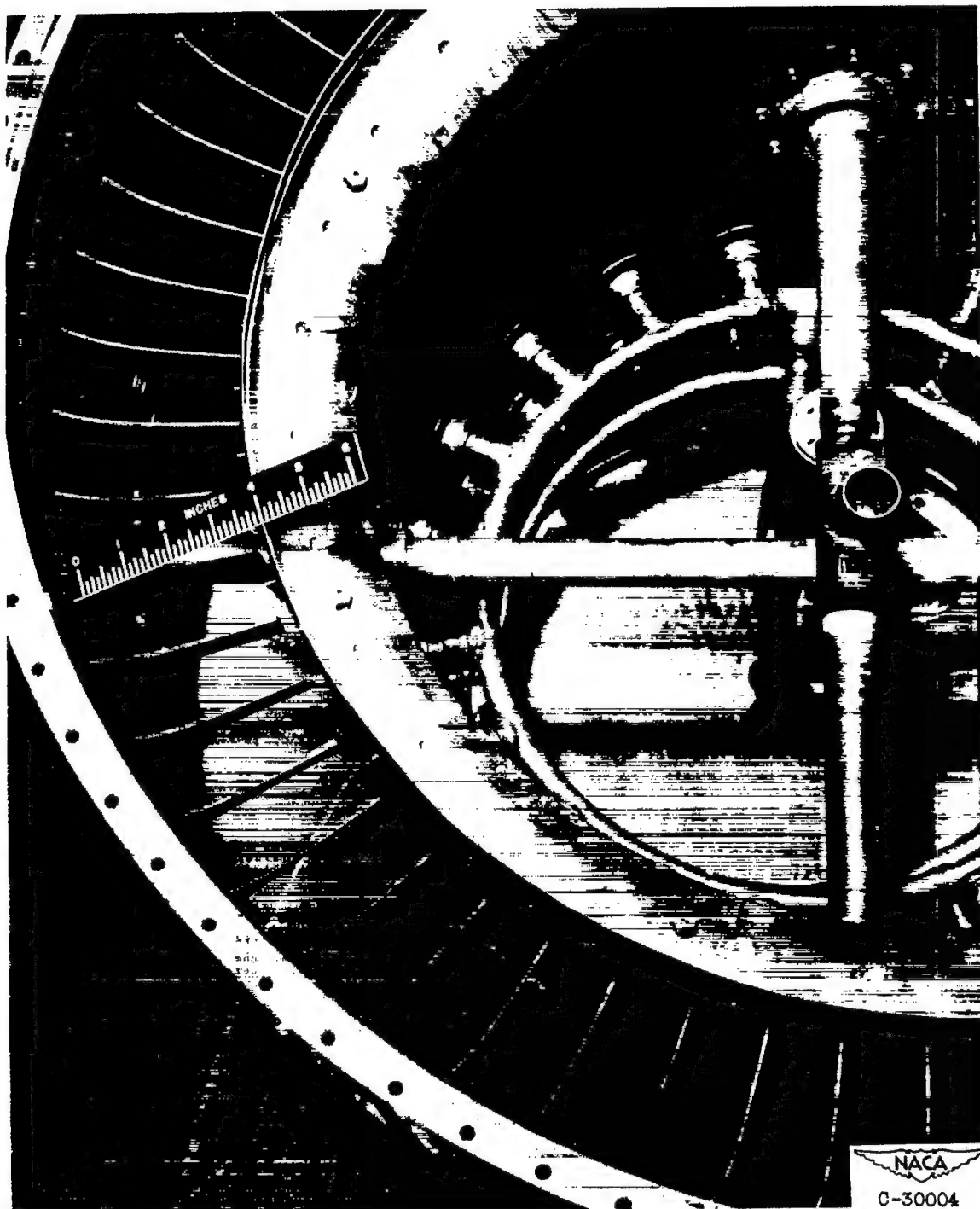


Figure 4. - Turbine-outlet gas-straightening vanes, vortex generators, and internal manifolds for configurations B, C, D, and E, as viewed in downstream direction.

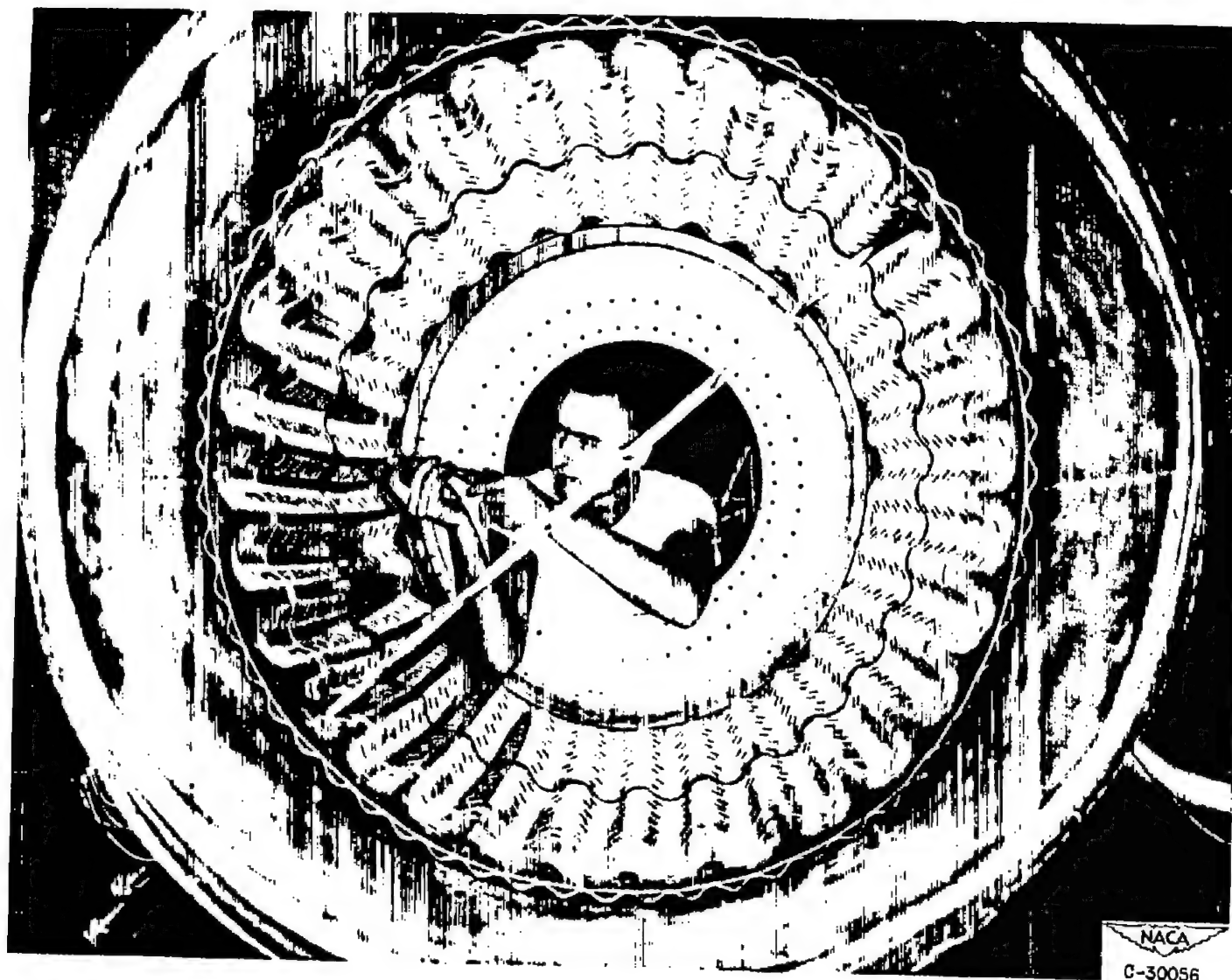
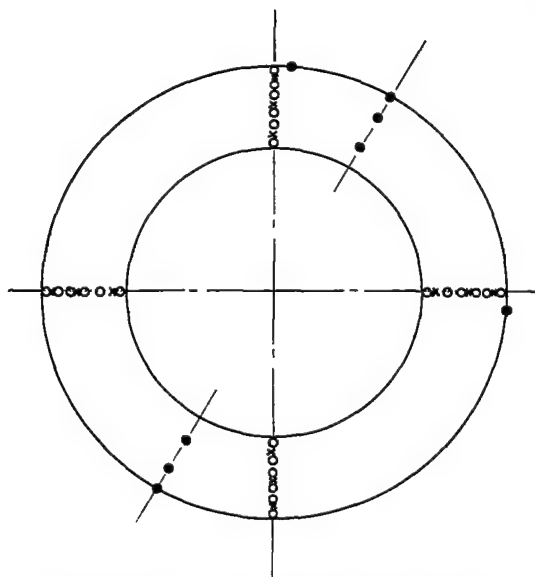
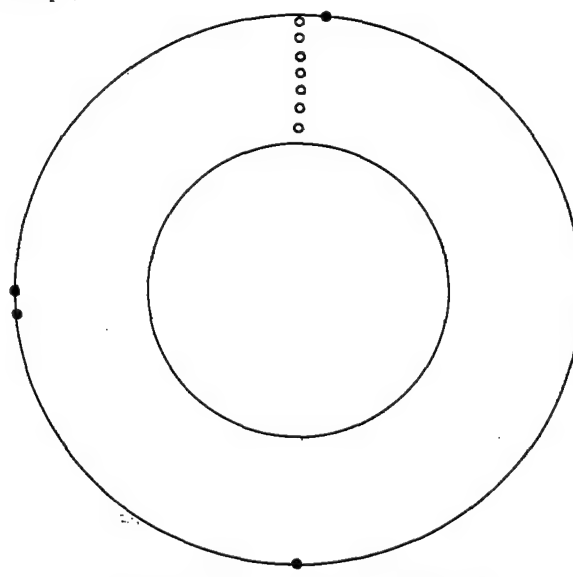


Figure 5. - Ceramic-coated corrugated liner installed in afterburner for configurations C, D, and E, as viewed in upstream direction.

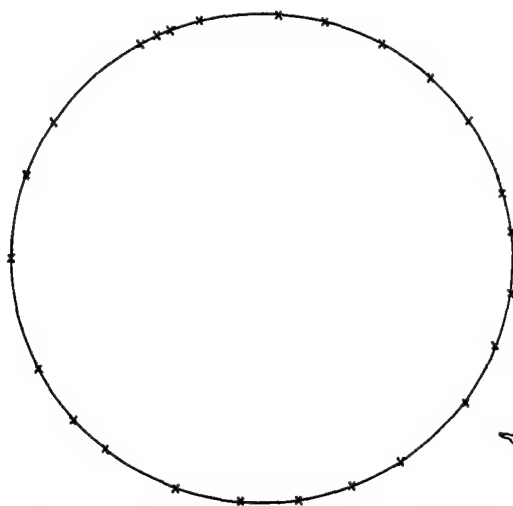
- Static-pressure tube (wall or rake)
- Total-pressure tube
- x Thermocouple



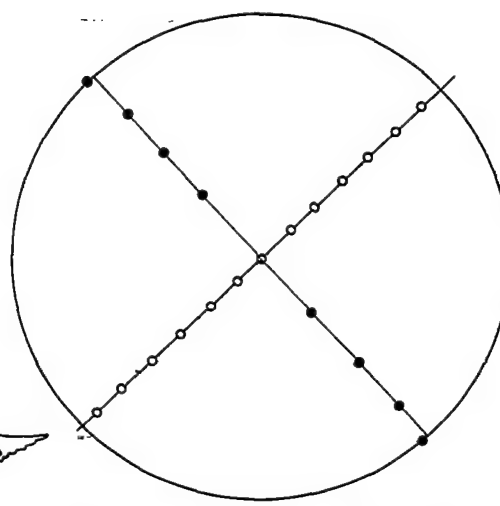
(a) Engine inlet, station 1, 21 inches upstream of leading edge of compressor-inlet guide vanes.



(b) Diffuser outlet, station 6,  $37\frac{13}{16}$  inches downstream of turbine-outlet flange.



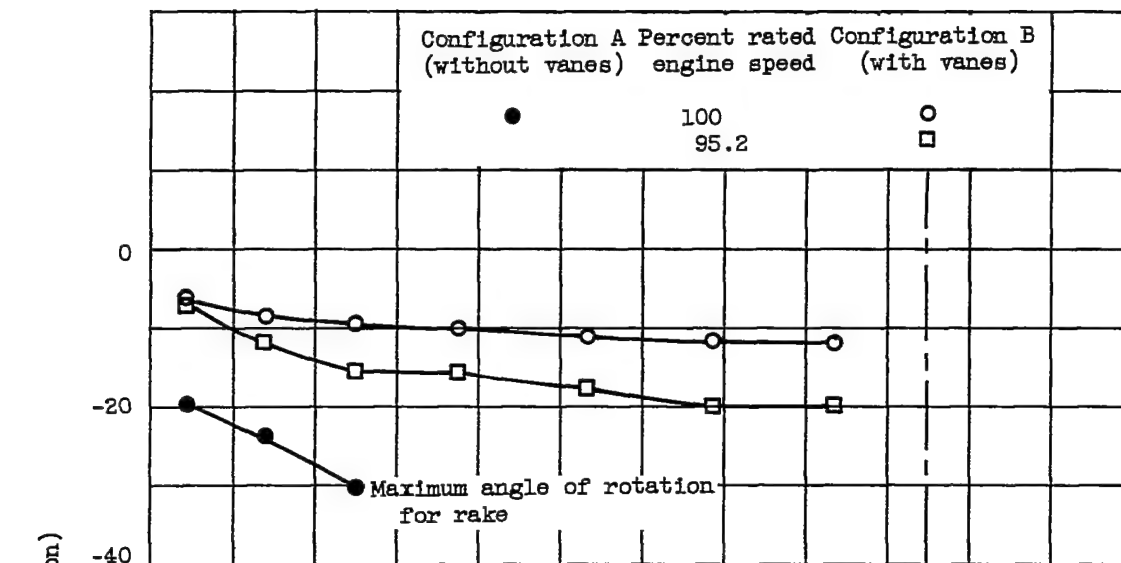
(c) Afterburner outer shell, station 8,  $79\frac{5}{8}$  inches downstream of turbine-outlet flange.



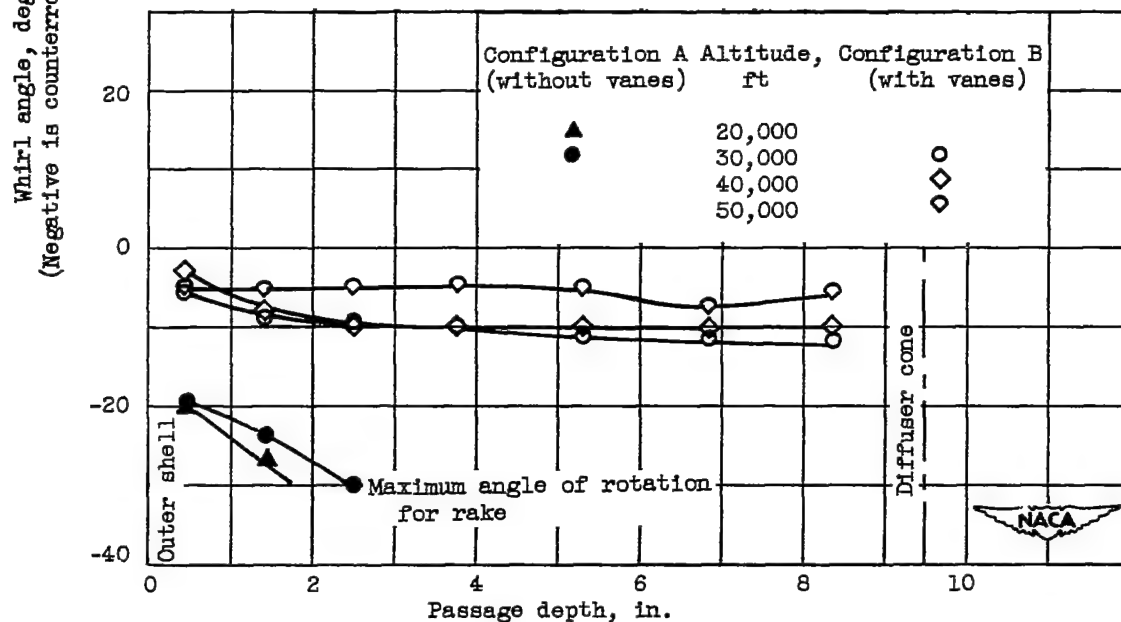
(d) Nozzle inlet, station 9,  $90\frac{3}{4}$  inches downstream of turbine-outlet flange.



Figure 6. - Instrumentation.



(a) Varying engine speed; altitude, 30,000 feet.



(b) Varying altitude; 100-percent rated engine speed.

Figure 7. - Effect of engine speed and altitude on turbine-outlet gas whirl at flight Mach number of 0.6.

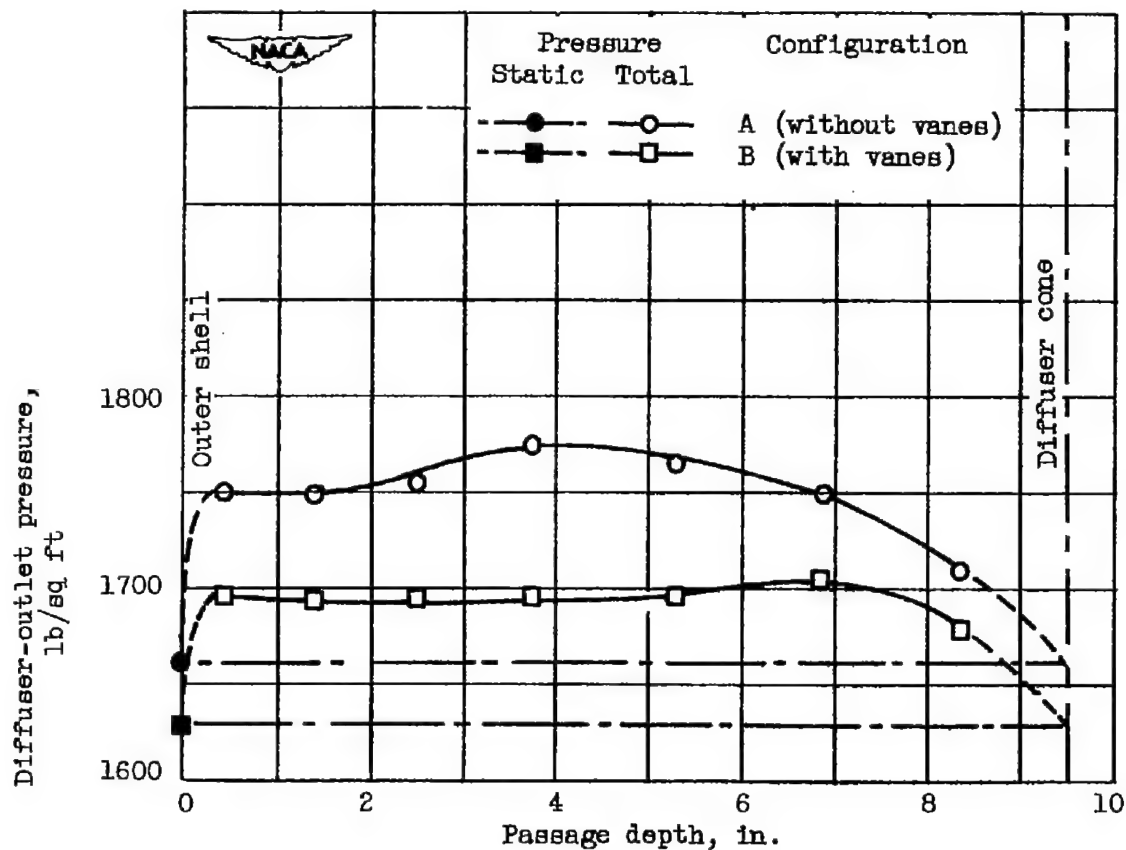


Figure 8. - Effect of turbine-outlet straightening vanes and vortex generators on diffuser-outlet total-pressure profile. Altitude, 30,000 feet; flight Mach number, 0.6.

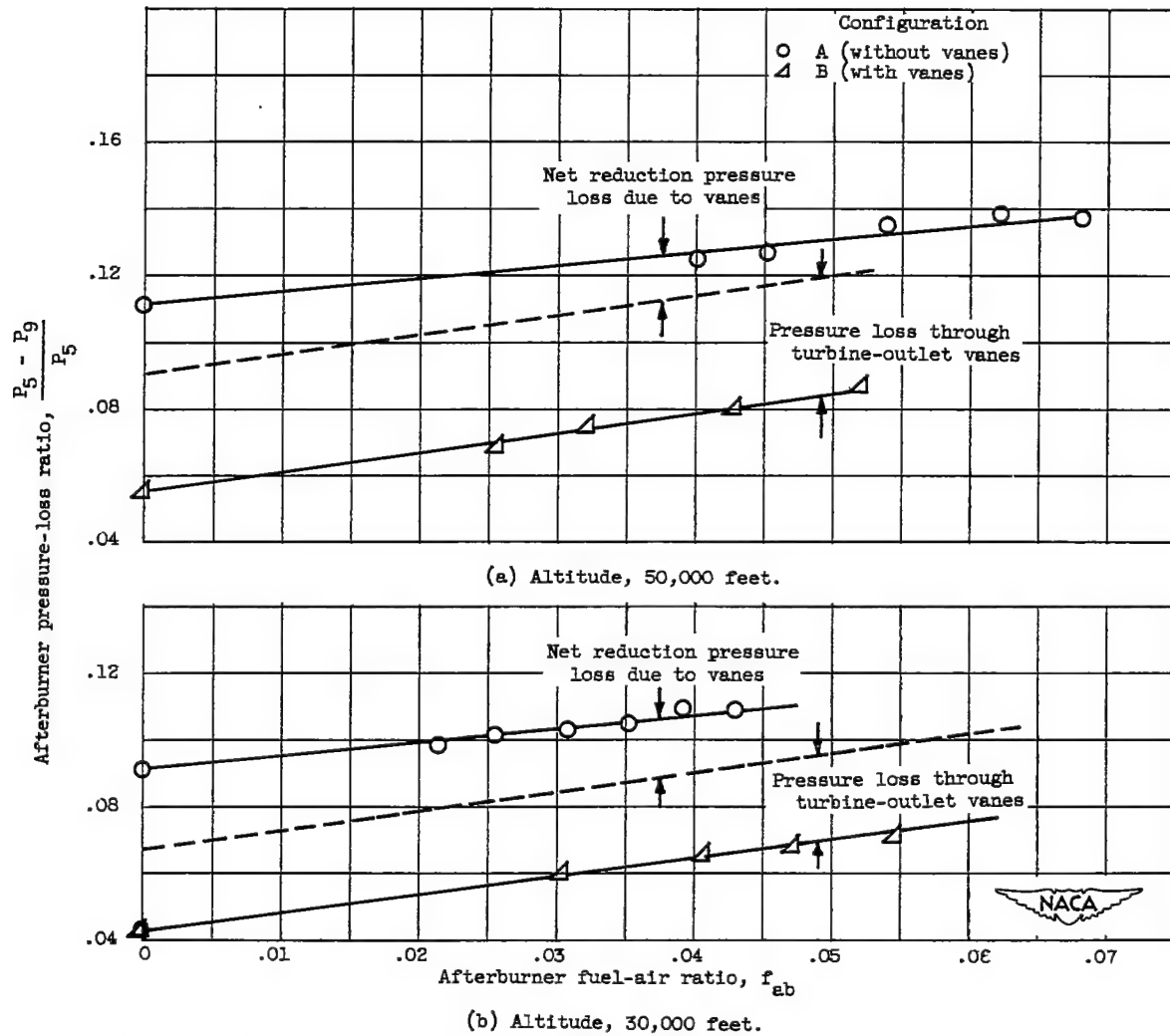
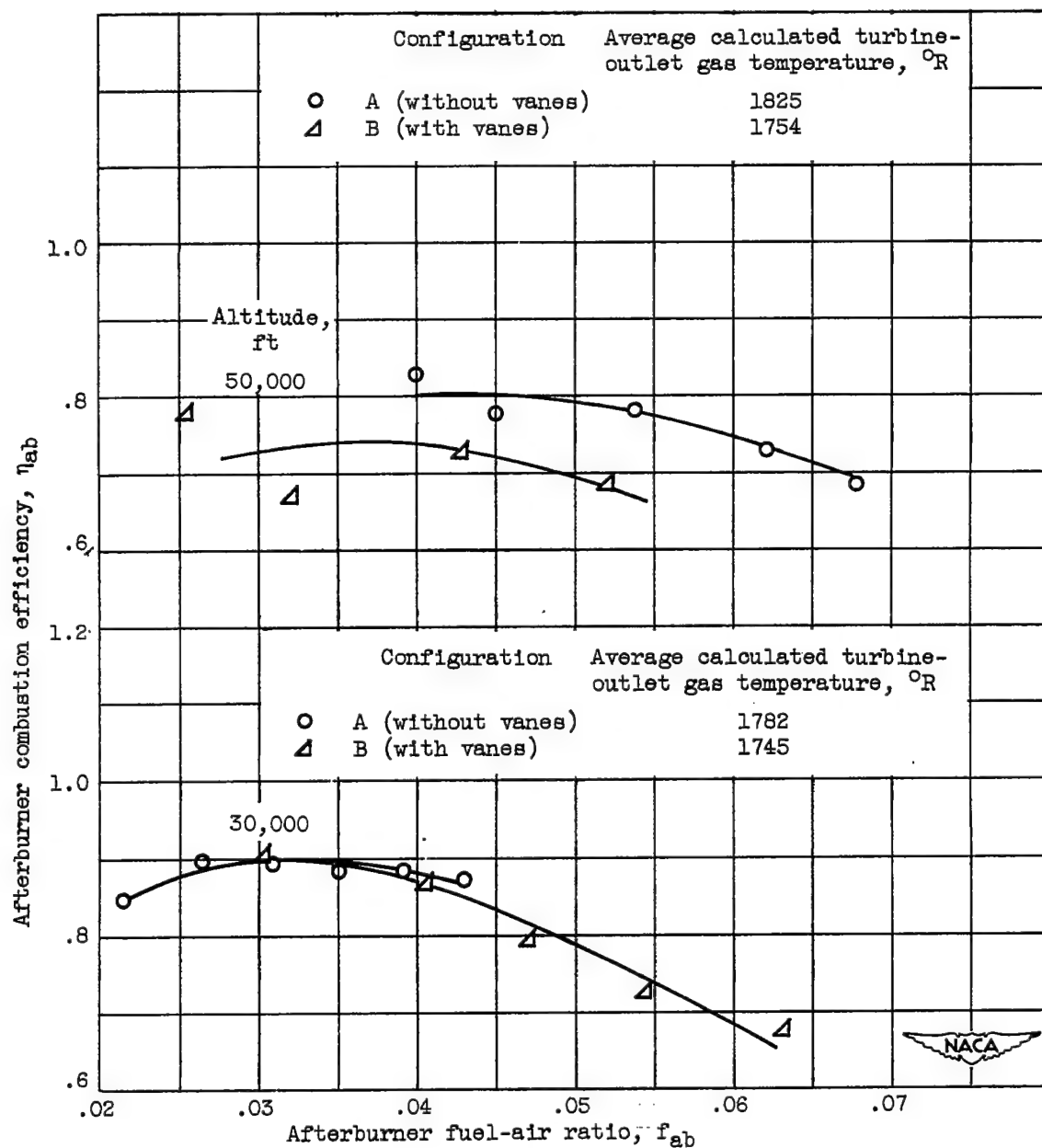
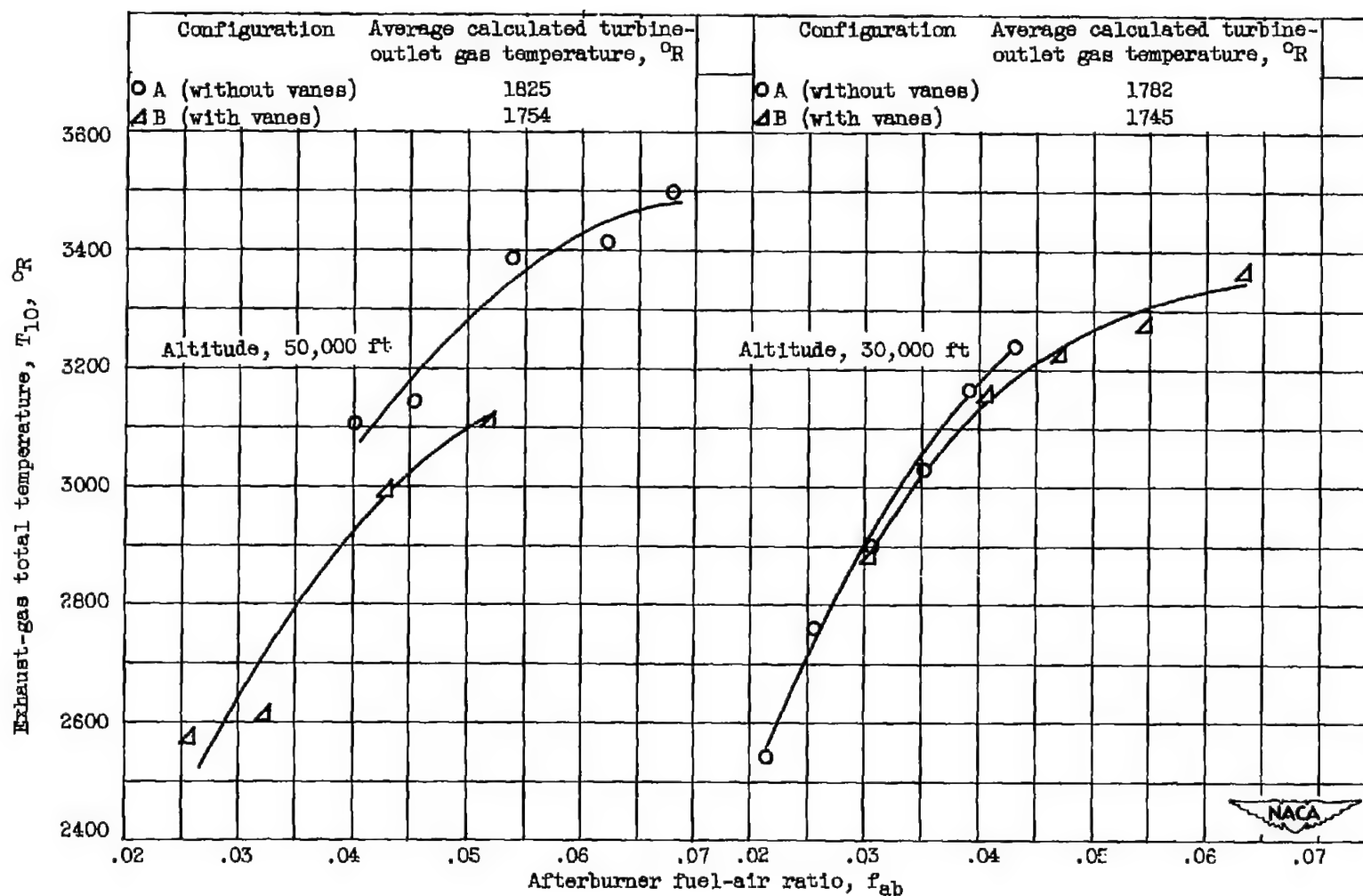


Figure 9 - Effect of turbine-outlet gas-straightening vanes, vortex generators, and corrugated liner on afterburner pressure loss. Flight Mach number, 0.6.



(a) Afterburner combustion efficiency (equivalent radial fuel distribution).

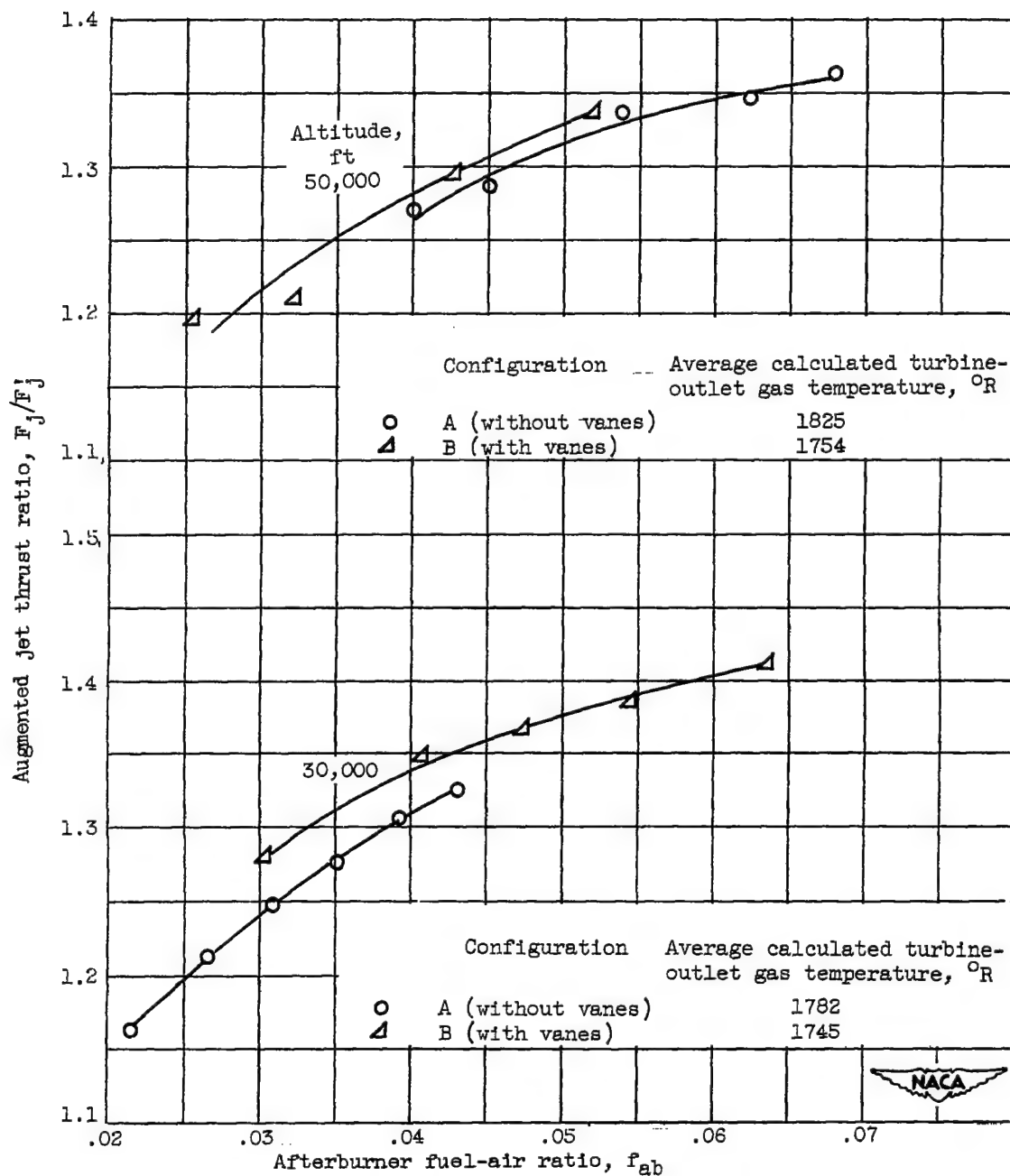
Figure 10. - Effect of turbine-outlet straightening vanes and vortex generators. Flight Mach number, 0.6.



(b) Exhaust-gas total temperature (equivalent radial fuel distribution).

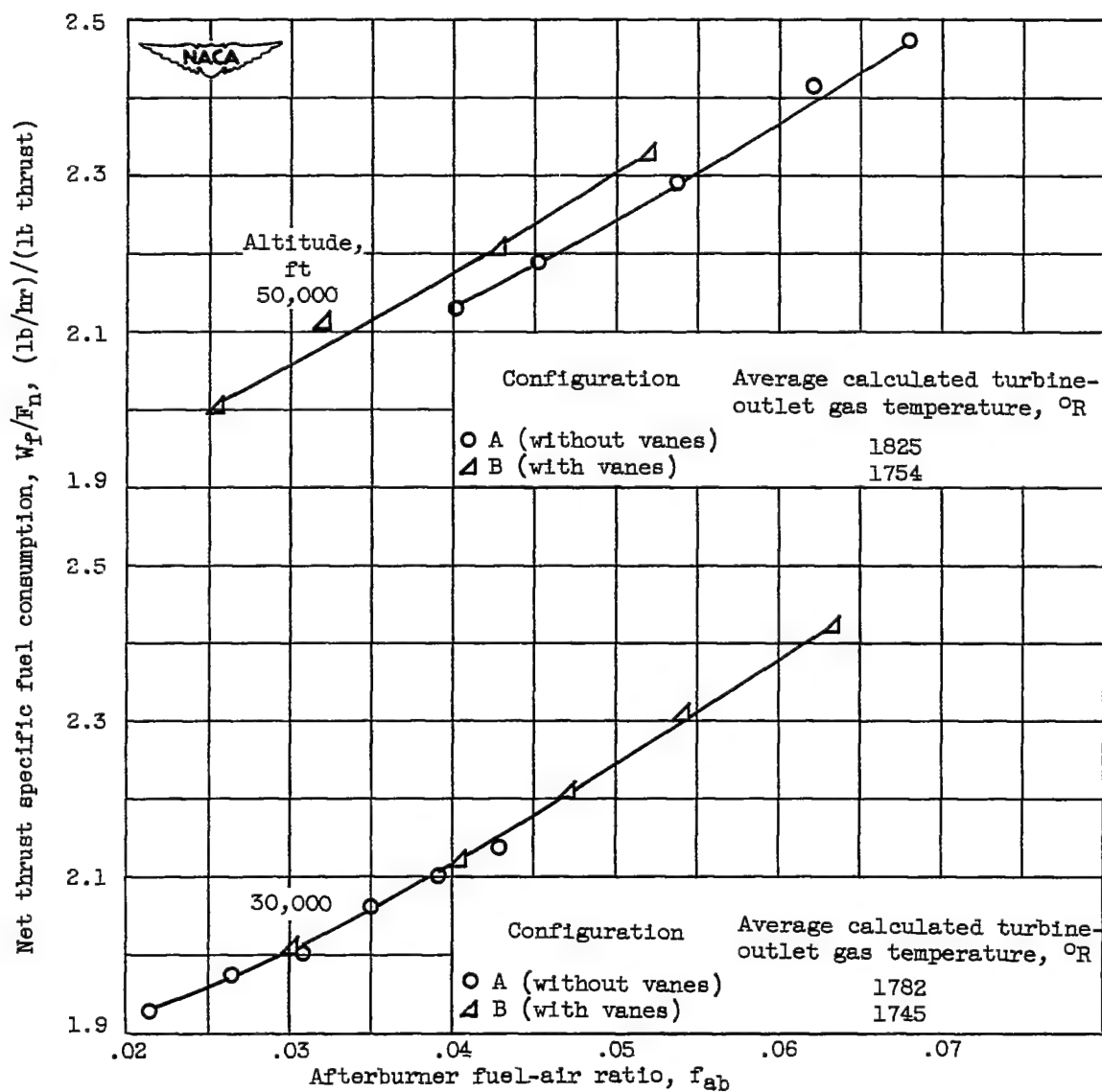
Figure 10. - Continued. Effect of turbine-outlet straightening vanes and vortex generators. Flight Mach number, 0.6.





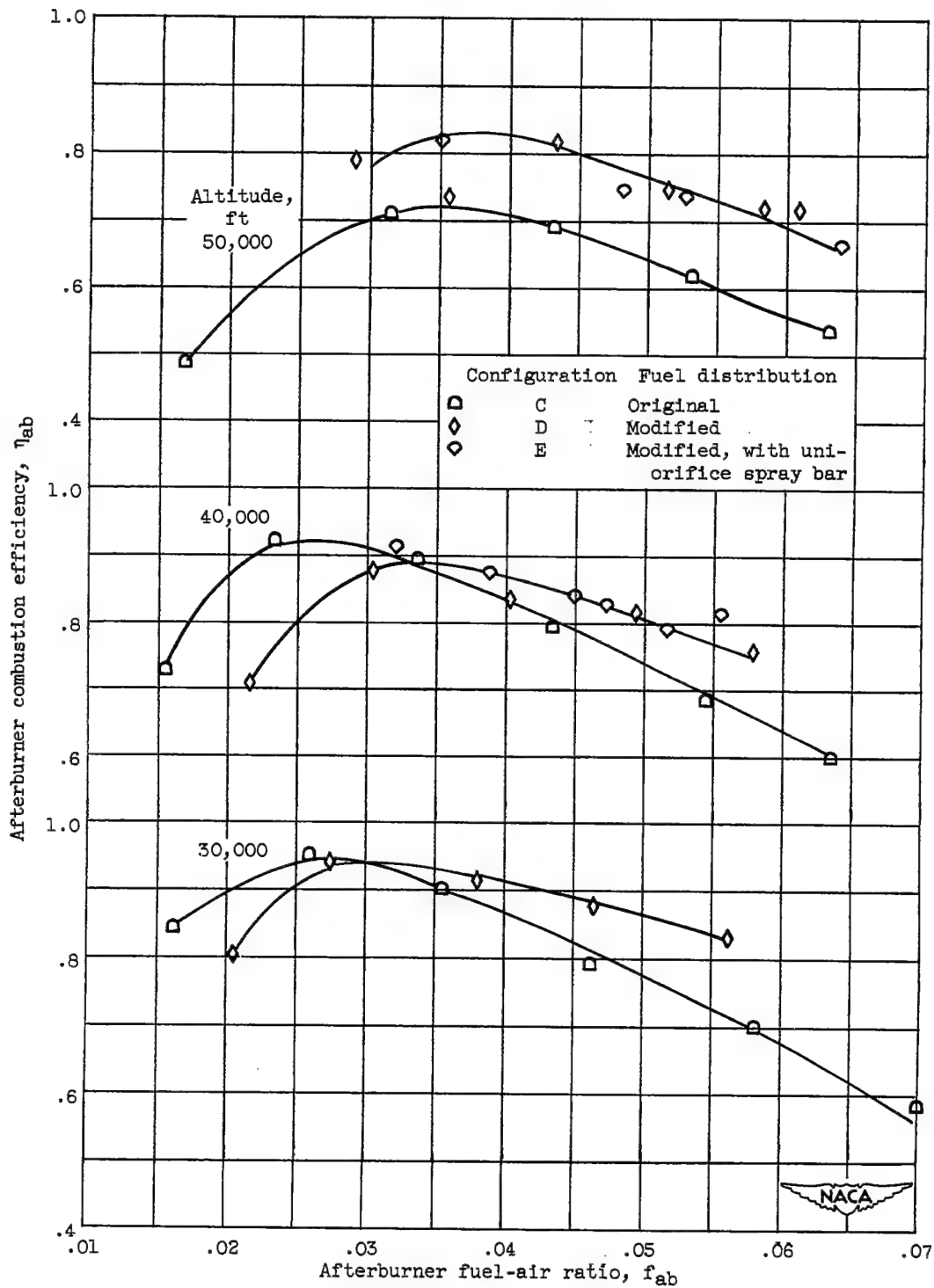
(c) Augmented jet thrust ratio (equivalent radial fuel distribution).

Figure 10. - Continued. Effect of turbine-outlet straightening vanes and vortex generators. Flight Mach number, 0.6.



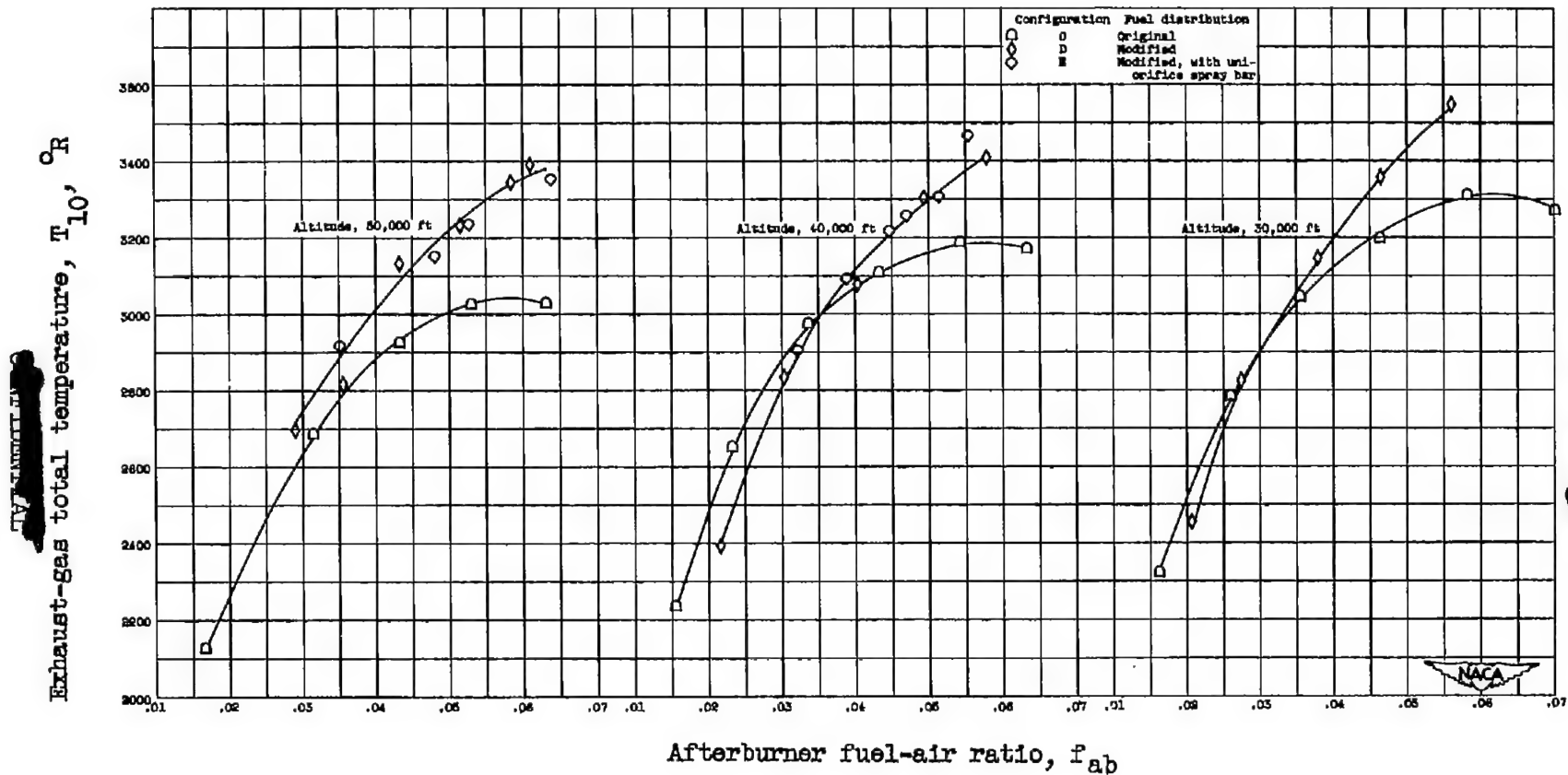
(d) Net thrust specific fuel consumption (equivalent radial fuel distribution).

Figure 10. - Concluded. Effect of turbine-outlet straightening vanes and vortex generators. Flight Mach number, 0.6.



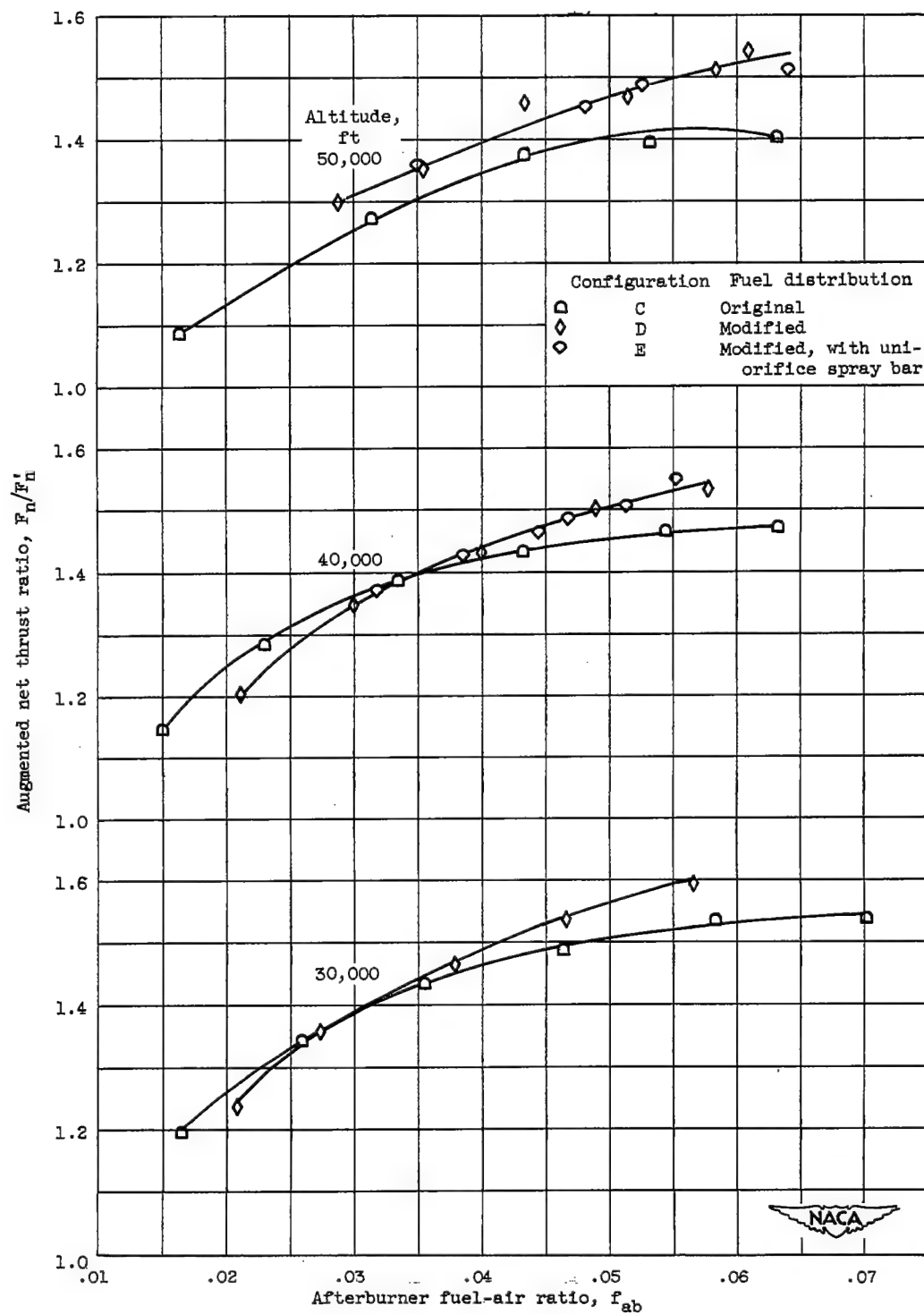
(a) Afterburner combustion efficiency.

Figure 11. - Effect of fuel distribution. Flight Mach number, 0.6.



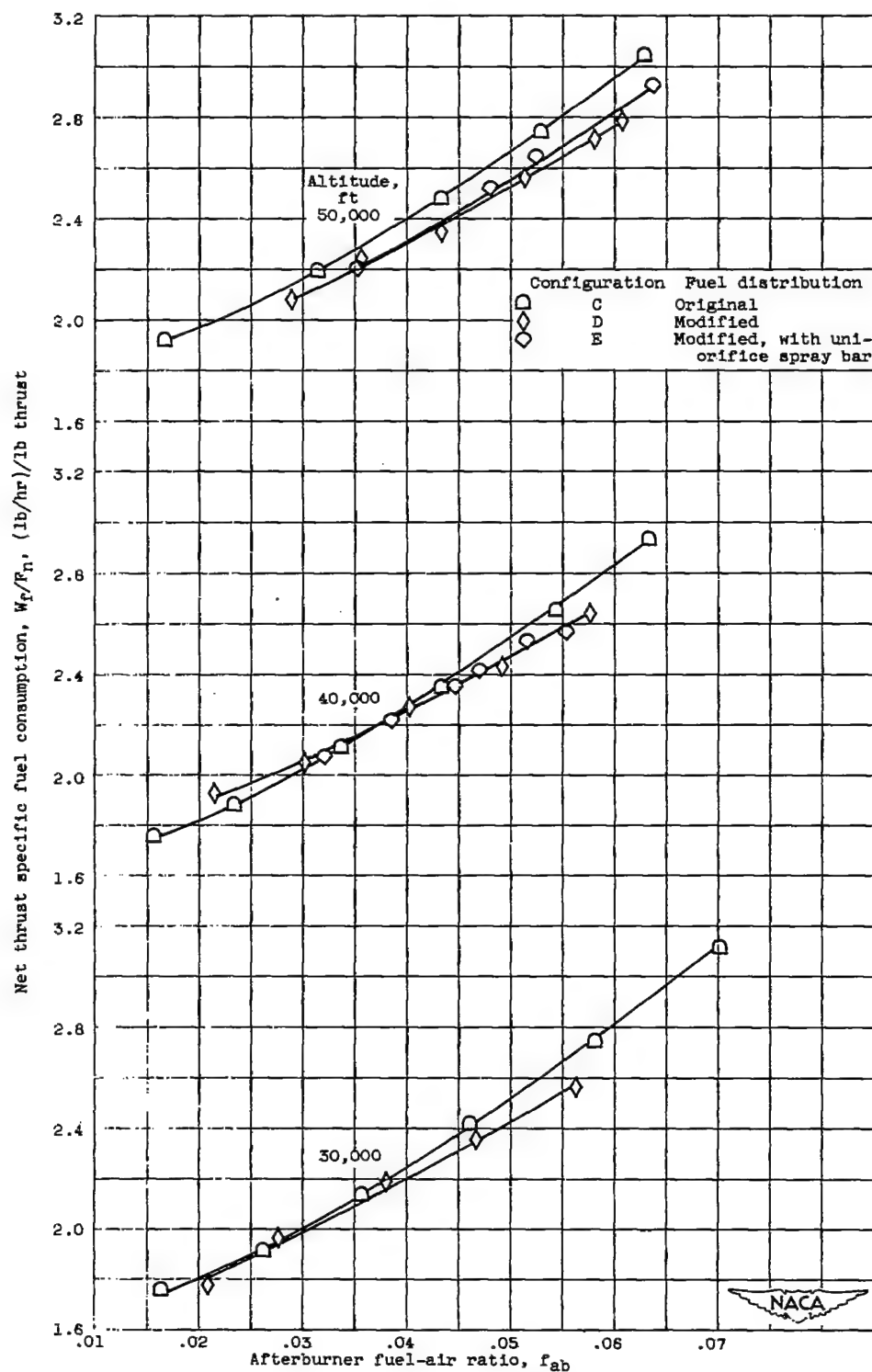
(b) Exhaust-gas total temperature.

Figure 11. - Continued. Effect of fuel distribution. Flight Mach number, 0.6.



(c) Augmented net thrust ratio.

Figure 11. - Continued. Effect of fuel distribution. Flight Mach number, 0.6.



(d) Net thrust specific fuel consumption.

Figure 11. - Concluded. Effect of fuel distribution. Flight Mach number, 0.6.

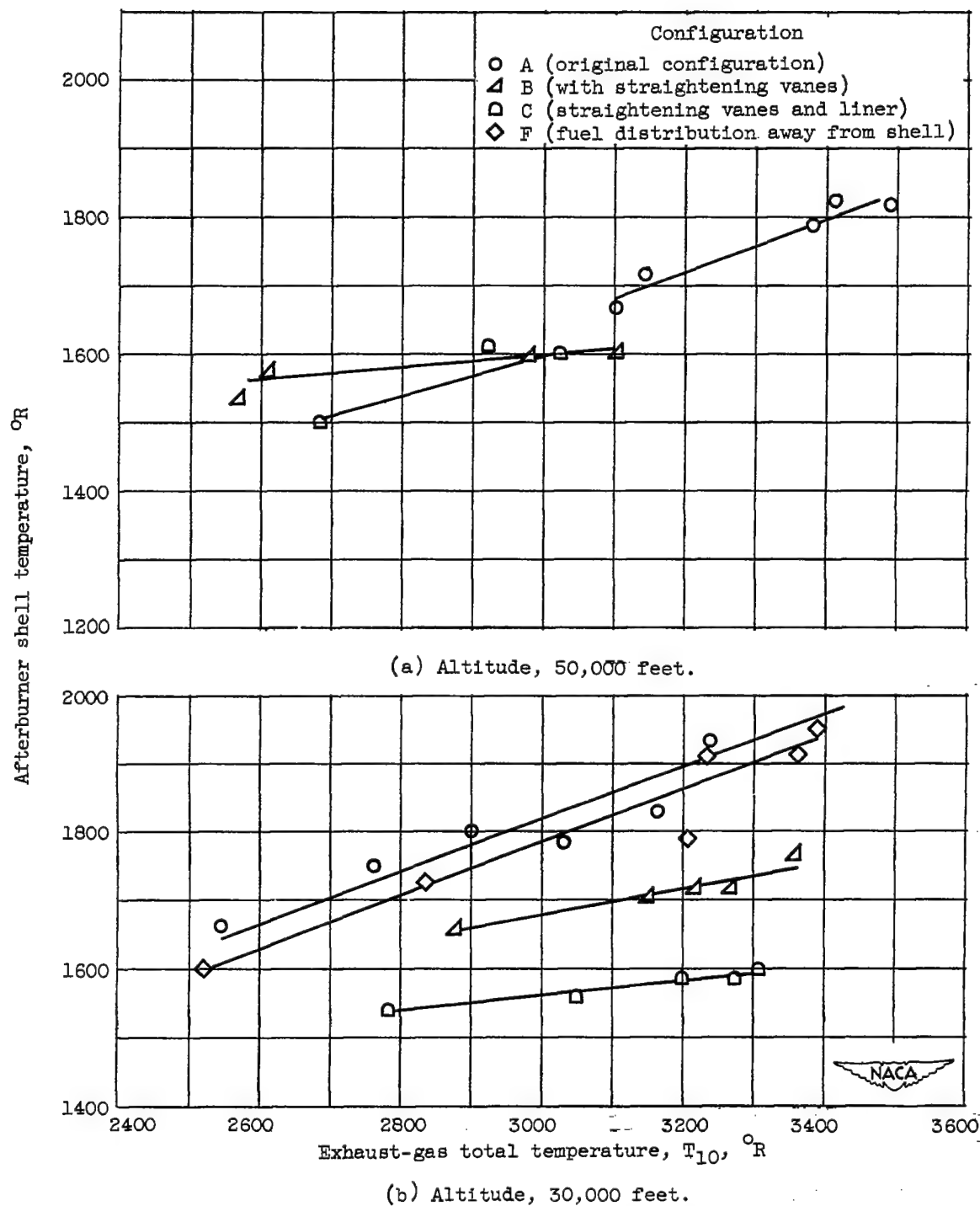


Figure 12. - Effect of turbine-outlet gas-straightening vanes and vortex generators, ceramic-coated corrugated liner, and fuel distribution on afterburner shell temperature. Flight Mach number, 0.6.

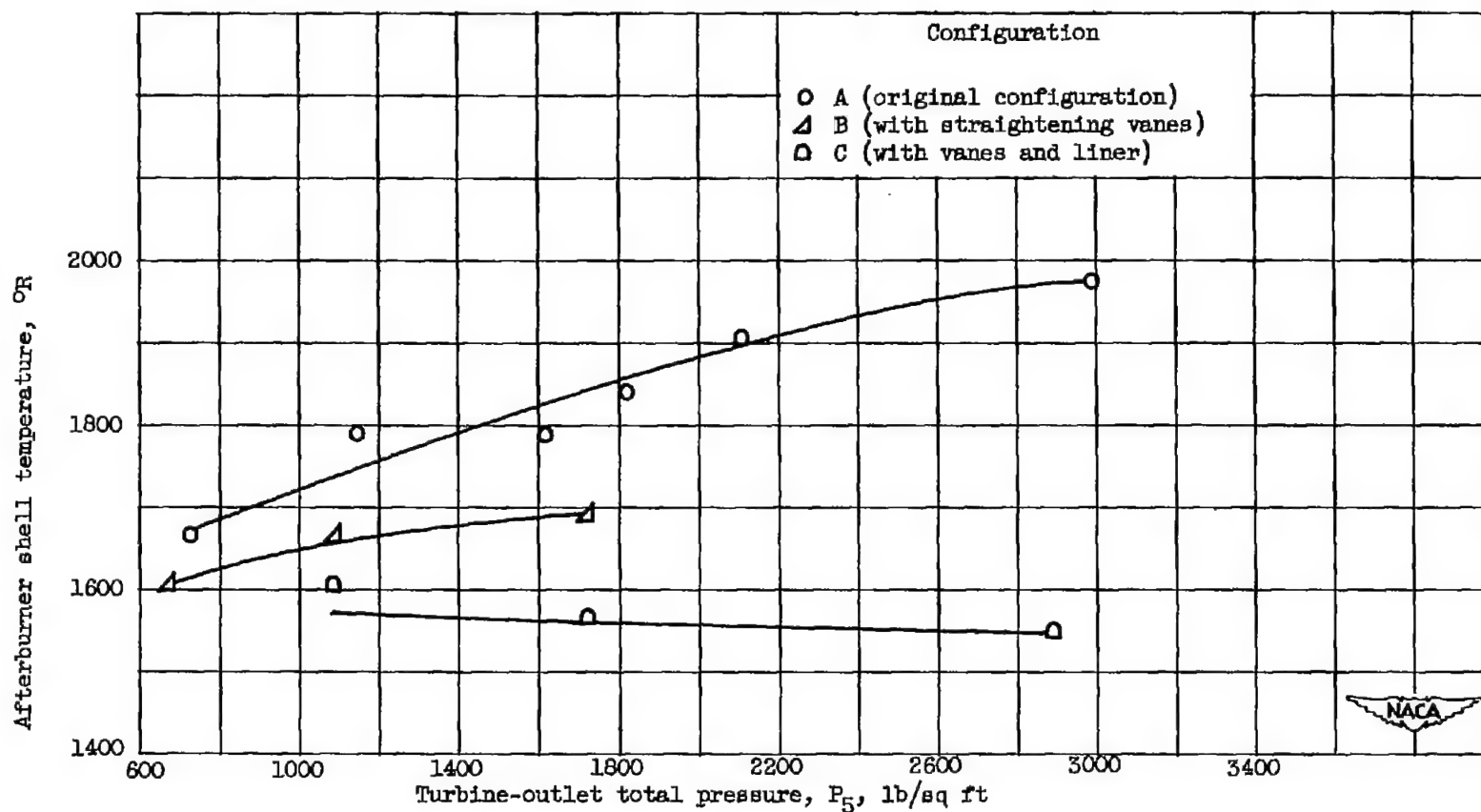
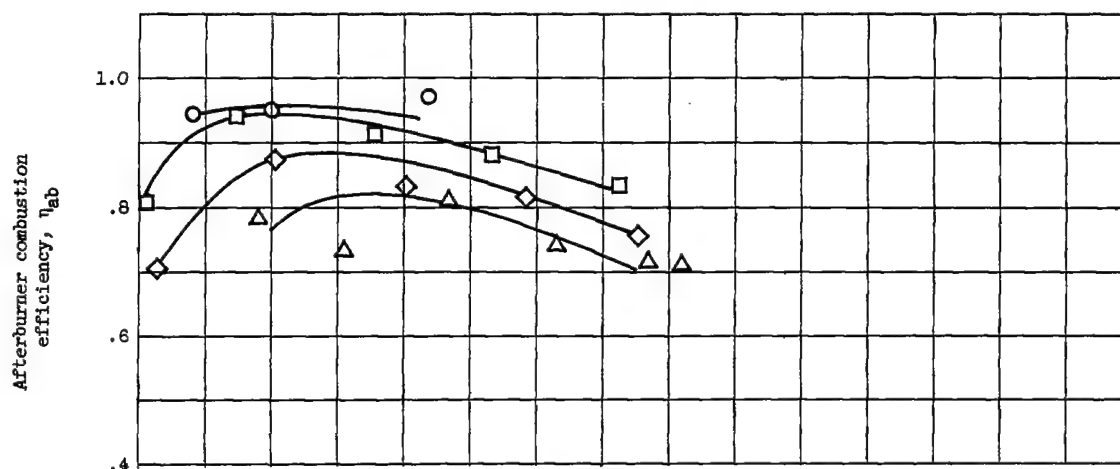
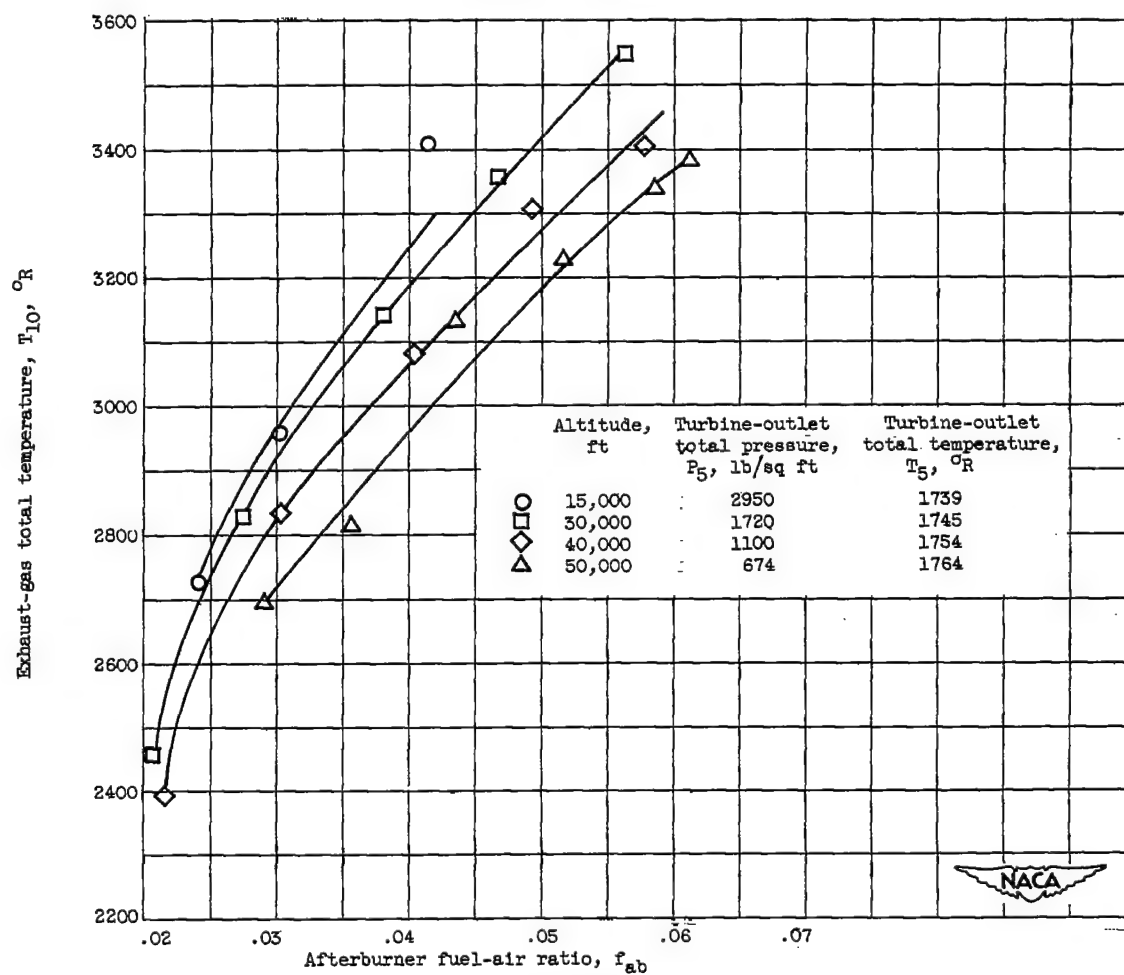


Figure 13. - Effect of turbine-outlet total pressure on afterburner shell temperature for gas temperature of 3100° R.



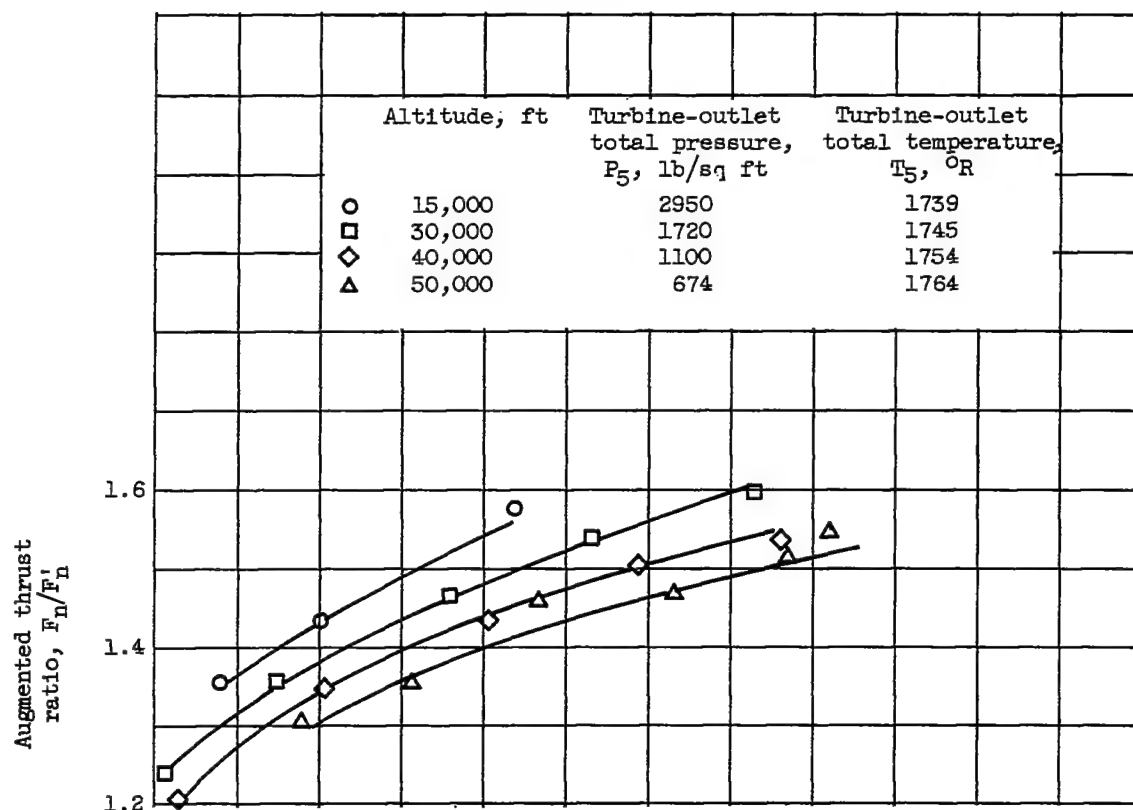


(a) Afterburner combustion efficiency.

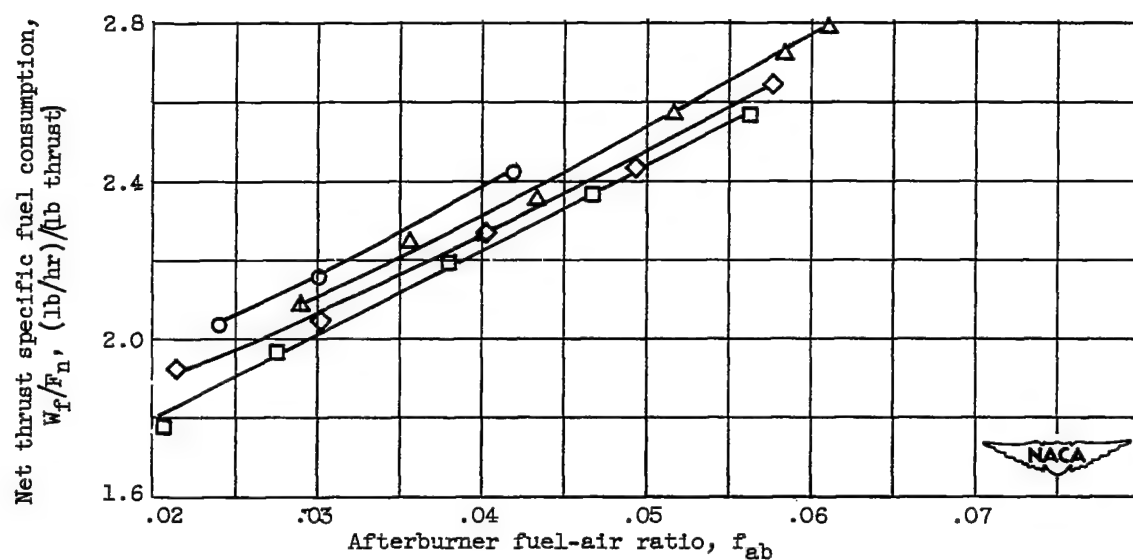


(b) Exhaust-gas total temperature.

Figure 14.- Effect of altitude on performance of configuration D. Flight Mach number, 0.6.

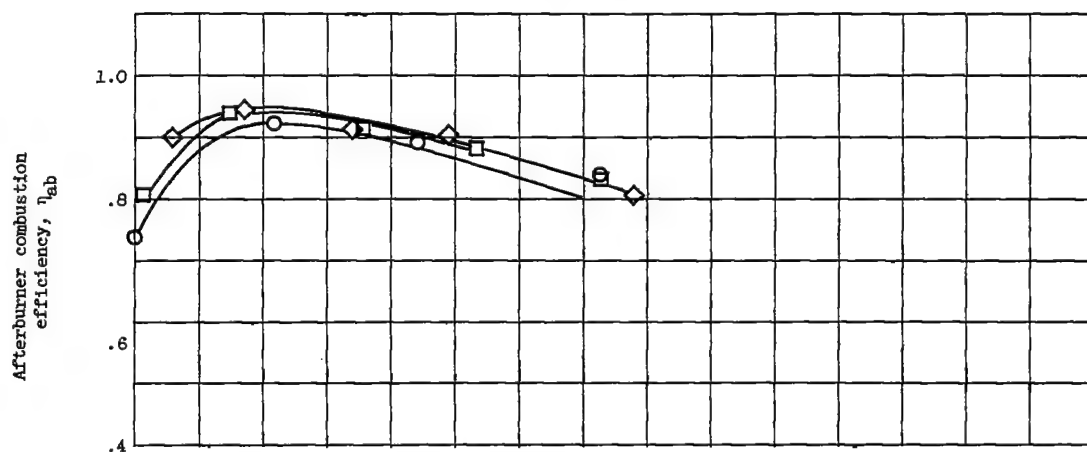


(c) Augmented thrust ratio.

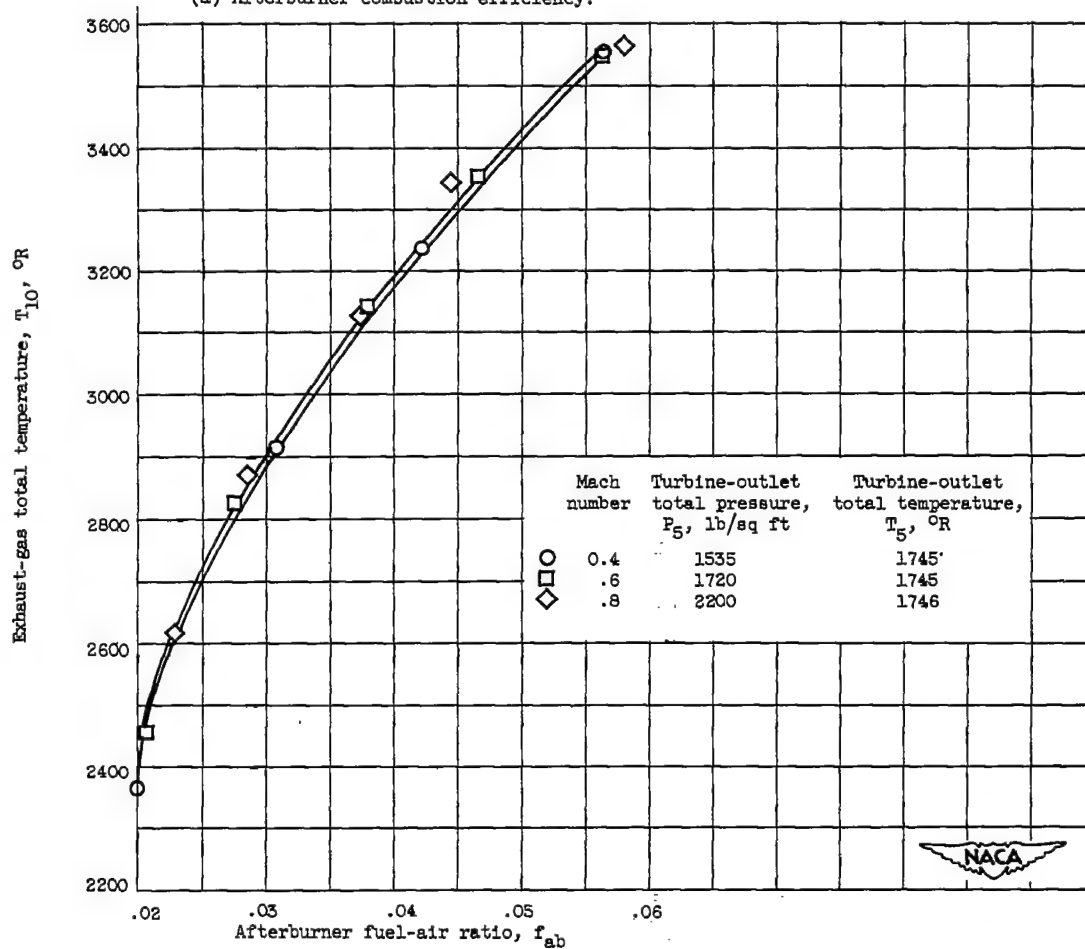


(d) Net thrust specific fuel consumption.

Figure 14. -- Concluded. Effect of altitude on performance of configuration D. Flight Mach number, 0.6.



(a) Afterburner combustion efficiency.

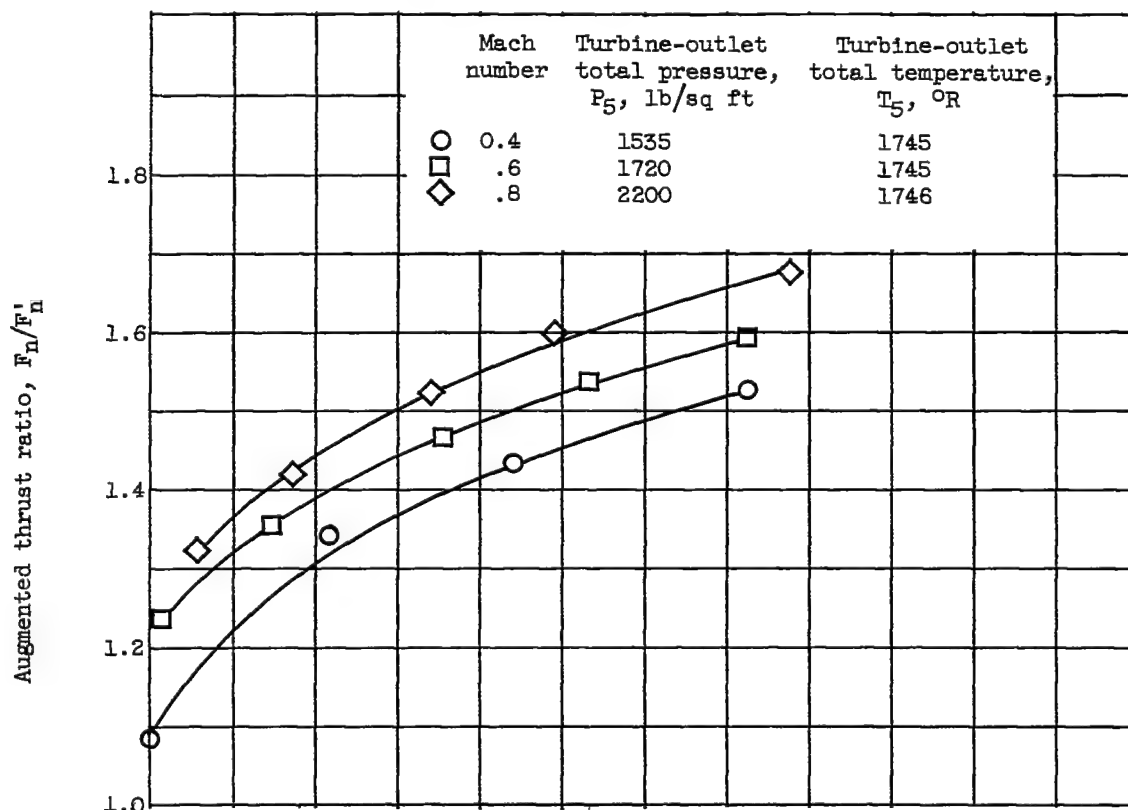


(b) Exhaust-gas total temperature.

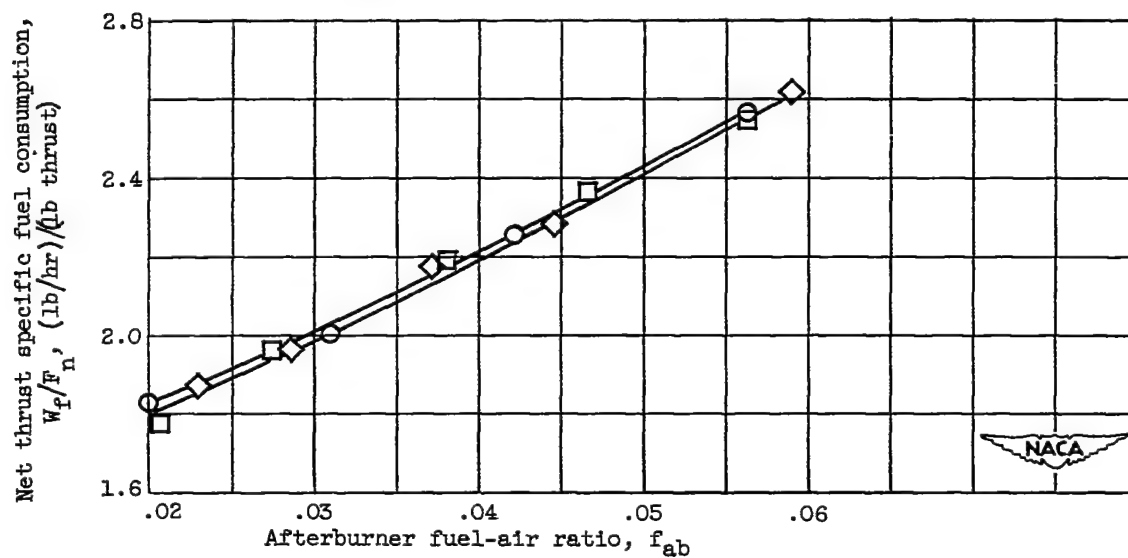
Figure 13. - Effect of flight Mach number on performance of configuration D. Altitude, 30,000 feet.

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(c) Augmented net thrust ratio.



(d) Net thrust specific fuel consumption.

Figure 15. - Concluded. Effect of flight Mach number on performance of configuration D. Altitude, 30,000 feet.

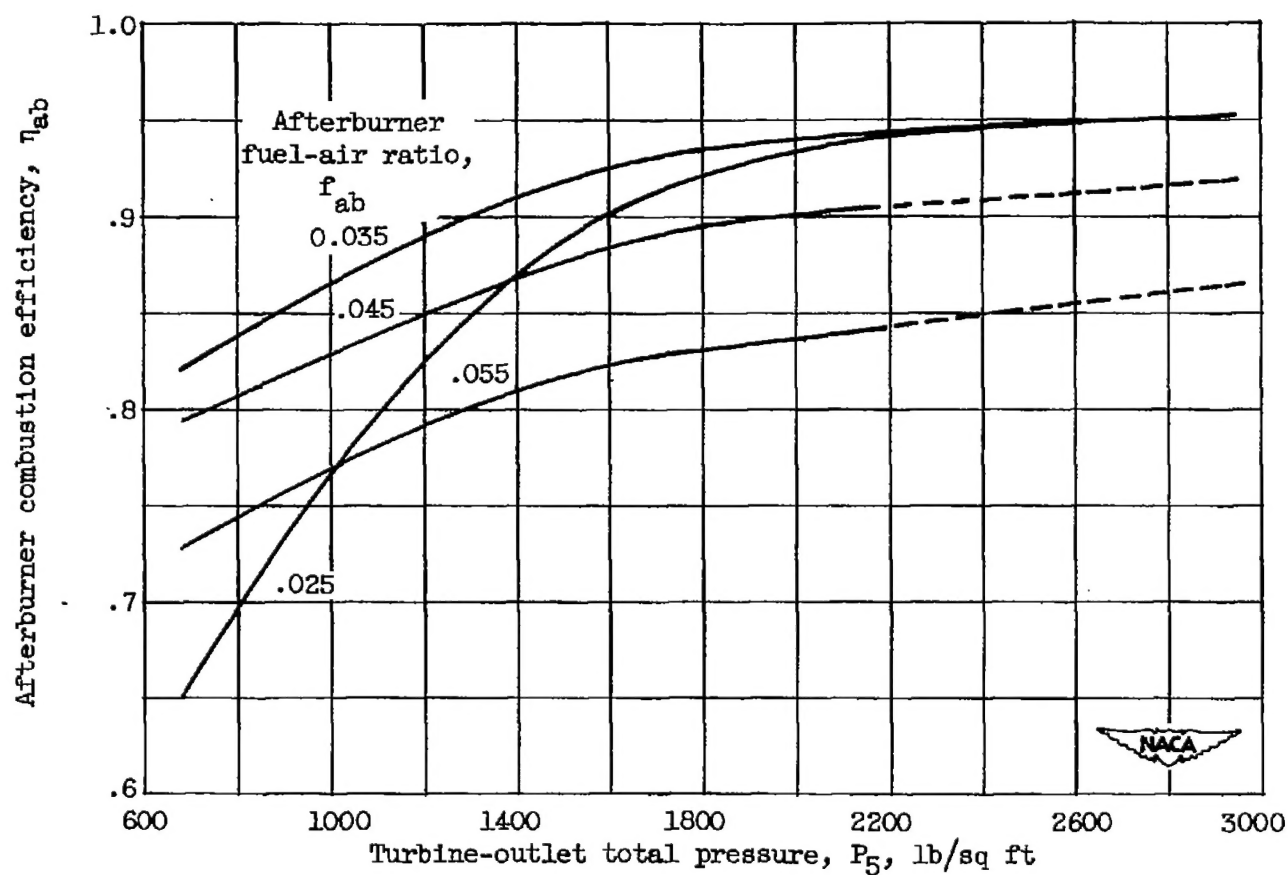
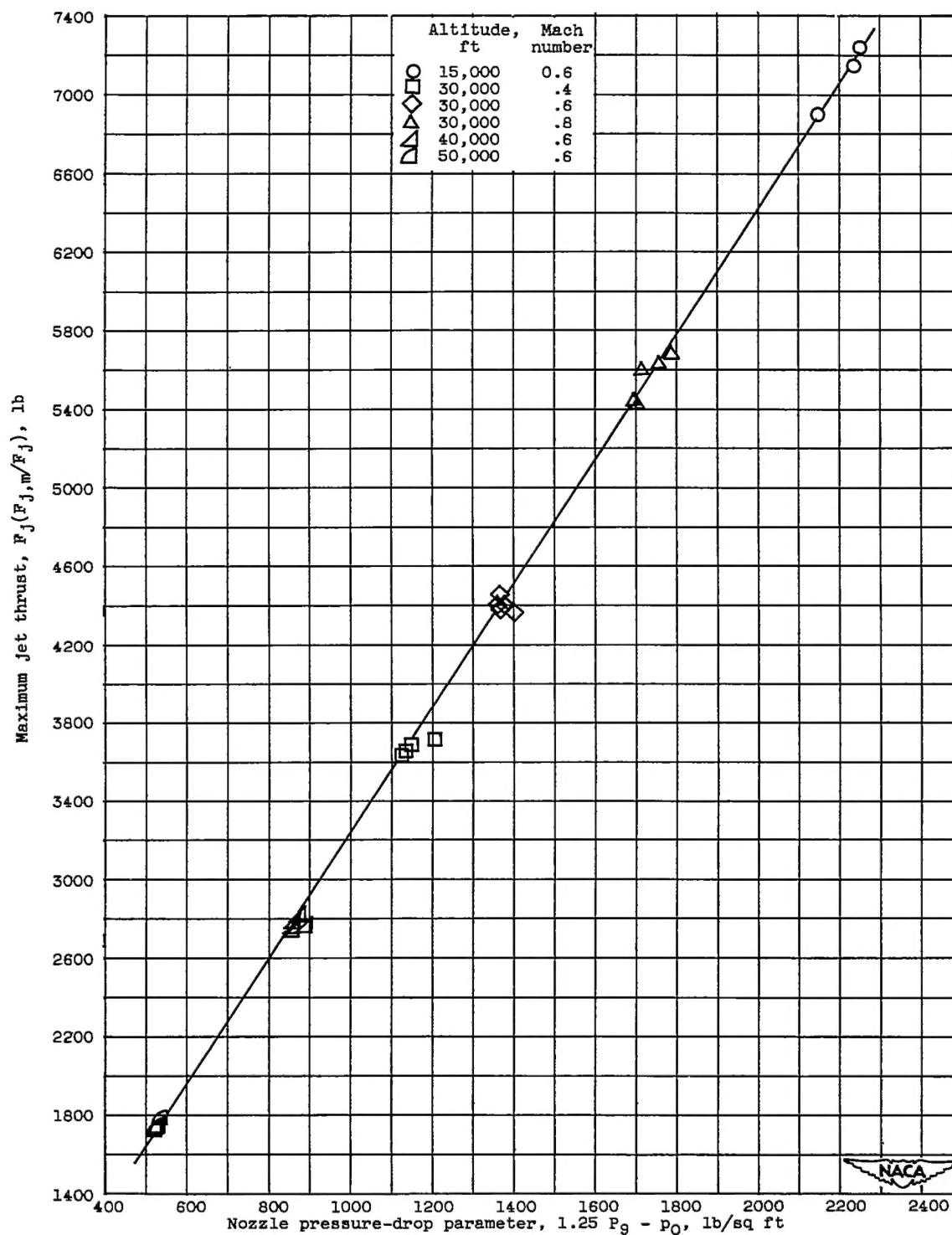
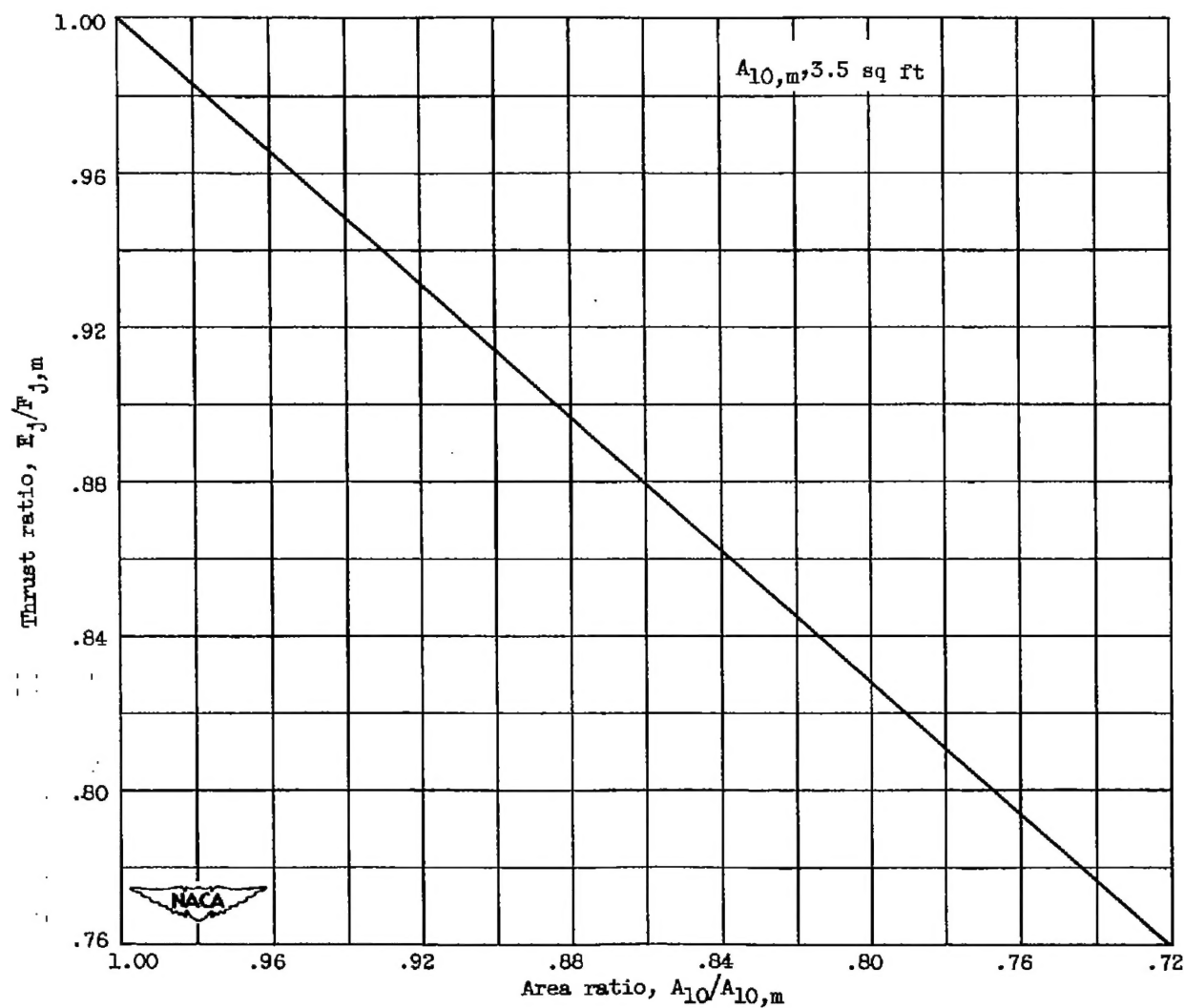


Figure 16. - Effect of turbine-outlet total pressure on combustion efficiency for various afterburner fuel-air ratios for configurations D and E.



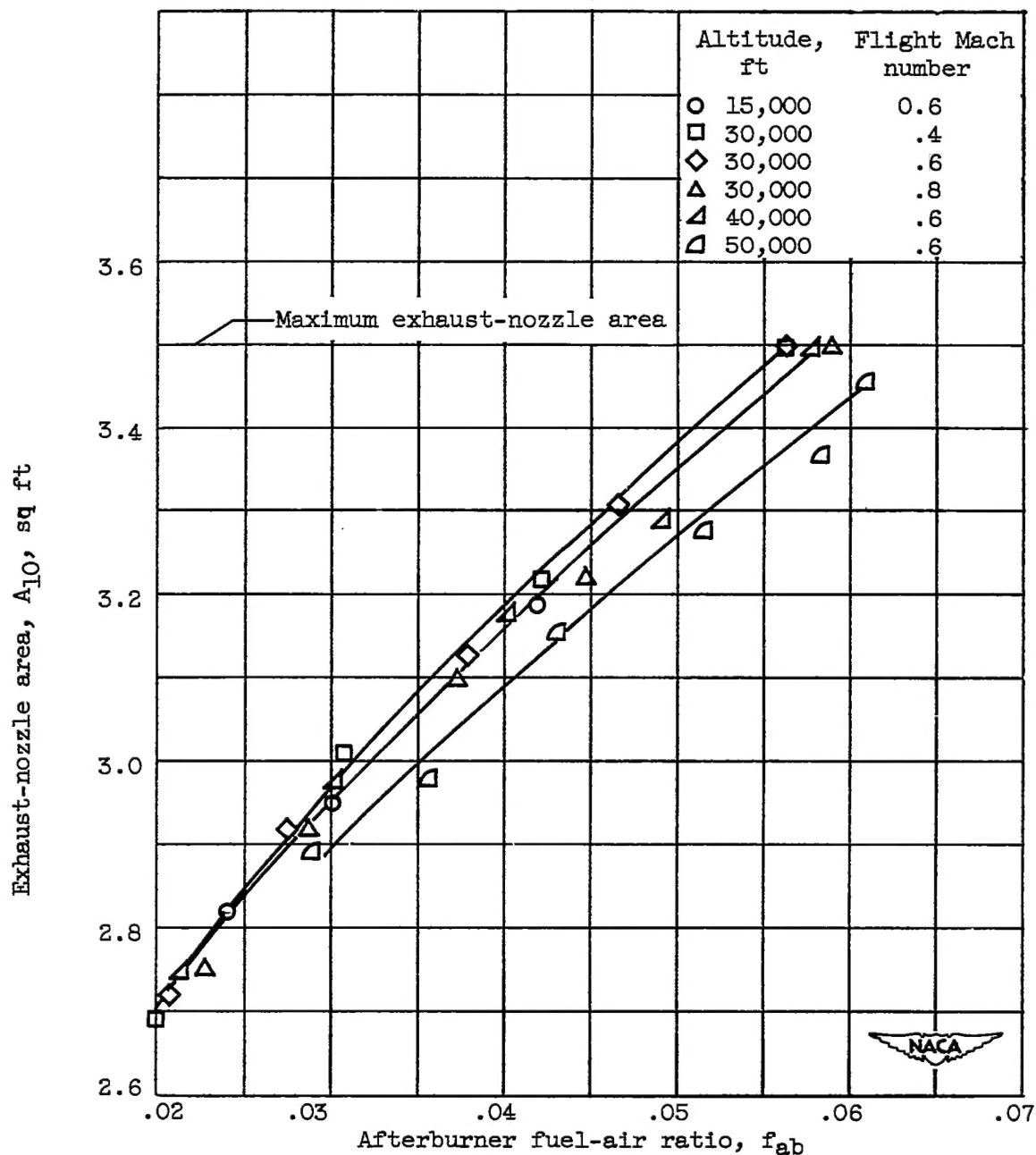
(a) Generalized jet thrust.

Figure 17. - Generalization of afterburning jet thrust for configuration D.



(b) Correction factor for jet thrust obtained from figure 17(a).

Figure 17. - Continued. Generalization of afterburning jet thrust for configuration D.



(c) Exhaust-nozzle area for various flight conditions.

Figure 17. - Concluded. Generalization of afterburning jet thrust for configuration D.